MEMORANDUM REPORT MS-1

STUDY OF AN EVOLUTIONARY INTERIM EARTH ORBIT PROGRAM

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ADVANCED CONCEPTS AND MISSIONS DIVISION
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SUMMARY

This report documents a study of a possible interim Earth orbital manned space flight program that would maintain continuous manned flights between the Skylab I mission and Space Station/Base operation. Although the Space Shuttle would become operational near the end of these interim missions, its impact upon this program was not evaluated. It considered an evolutionary, gradual, and step-wise spacecraft systems technology development from those as used on the Apollos and Skylab I to that required for the Space Station. The four mission spacecraft were dry workshop versions of the Saturn IV-B stage, and each would be individually configured, outfitted and launched by INT-21 vehicles. These spacecraft were evaluated for crews of three, six and nine men and for mission lifetimes of one year. Two versions of the Apollo CSM, a three man and a four man crew, were considered as the logistic vehicle. The Apollo CSM's would be inserted into orbit by either Titan III-M, Saturn I-B, or Solid Rocket Motored Saturn IV-B launch vehicles. Only the SRM Saturn IV-B vehicle can insert the crew, Apollo CSM and necessary logistics load with one launch.

A scientific plan was postulated for the program that could be completed during each spacecraft's mission by the size of crew available. This scientific program over the four missions could accomplish the equivalent of two years of experimental effort on the Space Station. A technical development plan for the life support and electrical power systems was so defined that first, the components would be flown as experiments, and then, they would be integrated into the later spacecraft as operating systems. The solar cell electrical power system of the first mission evolves into a light weight panel system supplemented by an operating isotope-Brayton system on the later missions. The open life support system of the first mission evolves to a system which recovers both water and oxygen on the last mission. The data handling, communications, radiation shielding, micrometeoroid protection, and orbit keeping systems were determined. The program costs were estimated and, excluding operational costs, the cost for each mission would average about $2 billion of which one-sixth would be for development, one-fourth for experiments, and the balance for vehicle acquisition.

This program as studied appears to be a viable interim alternative to continue manned Earth orbital flight should some events drastically change current NASA plans for the Space Shuttle and the Space Station. However, this program is only one of many alternative plans that should be similarly evaluated. It should be recognized that because it is an interim plan, it would use resources in its development that could not be recovered in the development either of the Space Shuttle or the Space Station/Base.

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INTRODUCTION

At this writing there are three manned space flight programs currently under study by NASA, i.e. the Earth Orbital Skylab I, the Space Station/ Space Base, and the Space Shuttle. There are many technology areas and operational capabilities required for the first Space Station which can only be acquired by flight experience. The Skylab I program will provide extensive flight experience; however, this will be limited and it appears reasonable to examine in detail a possible interim space station program between Skylab I and the Space Station to obtain even more experience and longer flight durations than would be available from Skylab I. The purpose of this report is to give the results of the study of one such interim space station program which capitalizes upon the investment which is being made in the development of Skylab I.

This study had a threefold purpose. The first was to delineate an Earth orbit flight program that adds to the operational and technology experience of Skylab I and prepares for the initial launch of the Space Station. The guidelines for such a program are as follows:

1. Develop mission control and flight operational experience under regular logistic supply flights.

2. Examine the different spacecraft systems for evolutionary development and in-flight certification.

3. Examine the implications of crew size and logistic flight requirements.

4. Try to maintain a relatively continuous manned flight capability.

The second purpose was to develop an experiment program that would recognize the status of the technology required as well as utilize the unique characteristics of Earth orbit flight. Such an experiment program would:

1. Use man's unique capability to perform the experiments.

2. Give prime priority to Earth oriented and Earth beneficial experiments.

3. Experimentally develop techniques to utilize the unique environment of Earth orbit for the Space Station program.

4. Define the limitations of the Earth orbit environment upon the experimental program.

The third purpose of this study was to evaluate the use of the dry Saturn IV-B stage as an interim Earth orbit spacecraft. Such an evaluation involves:

1. An assessment of the experiments to be performed, the experiment results, and the scientific accomplishments.

2. An assessment of the technology development gain for the Space Station program that could result from this plan, and finally,

3. An assessment of the program costs to permit comparisons of this plan for feasibility with other possible plans or program alternatives.

The interim program which was selected for study involves the use of four one-year life, dry SIV-B stations launched approximately 2 1/2 years apart. An Apollo CSM logistic system using 3 and 4-men versions of the command module was selected for logistic support. Saturn I-B, Titan III-M and a solid rocket SIV-B stage launch vehicles were selected for the logistic launches. It was recognized that the Space Shuttle would become available during the latter portion of this studied program; however, it was felt that its use and impact should be a separate consideration of this program if it proved to be viable as defined.

FLIGHT PROGRAM

Since the interim space station program would capitalize as much as possible on the use of hardware that has been developed primarily in the Skylab 1 program, it seems that the large engineering effort required for the systems and experiments in the Skylab I program should be utilized and amortized over more similar type missions provided that significant additional data could be obtained. Also, some minimum time is required between the end of one flight and the launch of the next to permit incorporation of the minor lessons learned from previous flights into the new vehicle. In this study, it has been assumed that the developments and experience resulting from Skylab I are utilized and that the flights are spaced so that a time of about 1 1/2 years occurs between the end of one space station use and the launch of the next vehicle.

The flight program that was formulated as the basis for this study is shown in Figure 1. This program had been evolved from examining the effects of a constant budgetary funding level of $3.3 billion upon the NASA flight programs, and the use of available spacecraft to maintain manned flight between the Skylab I and the Space Station/Shuttle flights This program consists of four distinct space stations launched at 2 1/2 year intervals. The first mission is assumed to start about 2 1/2 years after the Skylab I mission. Each mission requires a station lifetime of at least one year, and thus all systems must be qualified as reliable for this time period. Each space station is assumed to be left stored in space in a reactivable status at the completion of its mission. With the 2 1/2 year interval, a major modification, the need for which may become apparent on one space station, could be developed and engineered for the station which would follow the next space station launch. Figure 1 shows that the Apollo program as now configured has been assumed to terminate after the completion of the Apollo XV flight. Although the Space Shuttle is shown to start its flights during the flight of the later interim space stations, for simplicity it was not considered as a logistic vehicle for this study.

The first space station would carry a crew of three men continuously for one year. During this time the crew would grow to six men at the time of logistic resupply and crew change. The second space station would have an average crew of six men, and the third and the fourth stations would have crews of nine men. As an option to this program, an evaluation would be made of having four-man logistic craft to support the last two space stations. The sections which follow contain details of the spacecraft used, logistic support intervals and experiment programs.

Spacecraft

The assumption for this interim program that the Apollo program is terminated after Lunar Flight XV, means that four Saturn V class vehicles are not utilized on Apollo flights XVI through XIX; thus these vehicles may be used as the launch vehicles for the four interim space stations. These space stations would be configured as livable operational spacecraft derived from the Saturn SIV-B upper stages in the similar manner to that being

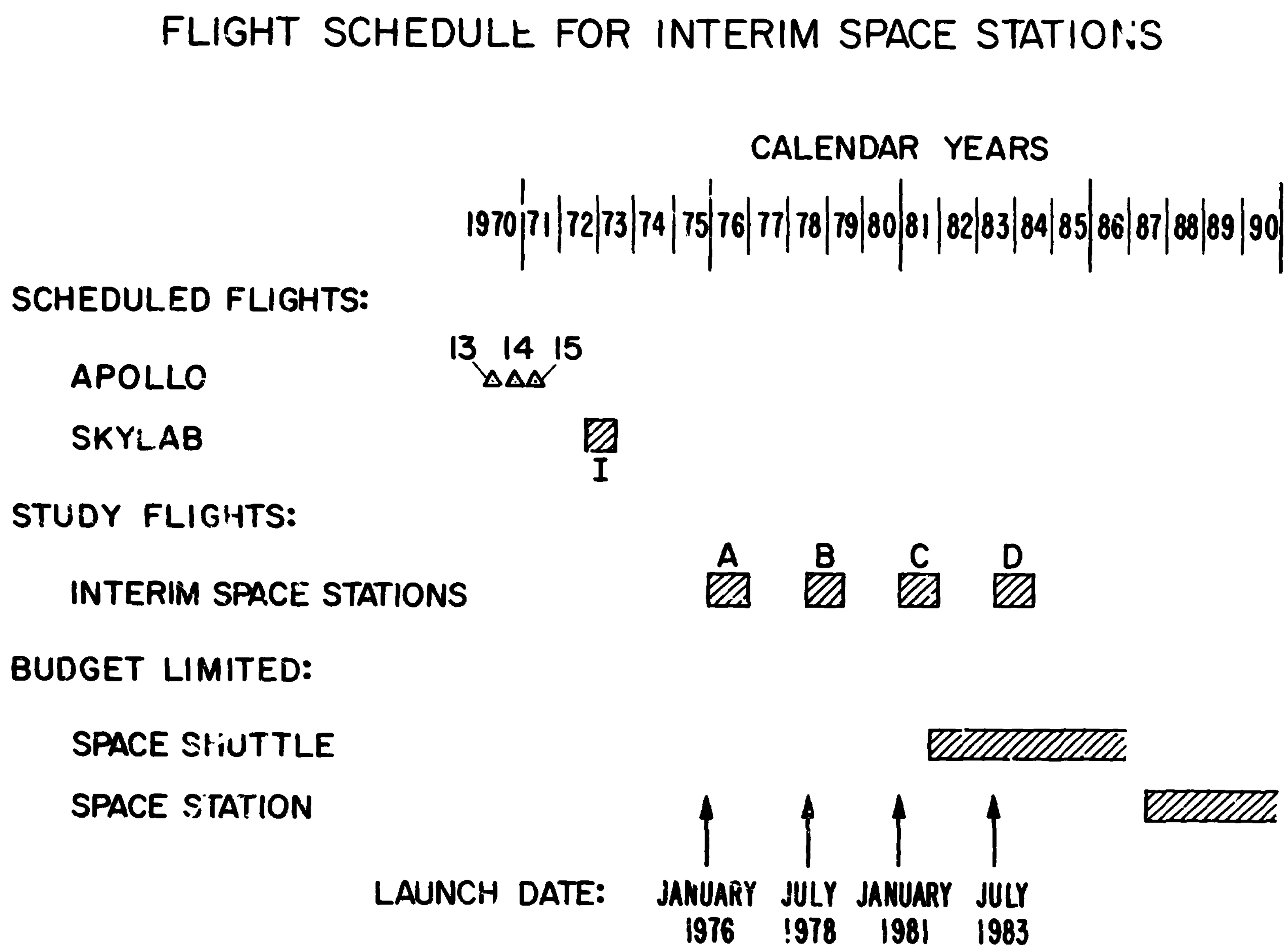
FLIGHT SCHEDULE FOR INTERIM SPACE STATIONS
CALENDAR YEARS
1970 71 72 73 74 75 76 77 78 79 80 81 82 83 84 85 86 87 88 89 90
SCHEDULED FLIGHTS:
APOLLO
13 14 15
SKYLAB
I
STUDY FLIGHTS:
INTERIM SPACE STATIONS
A
B
C
D
BUDGET LIMITED:
SPACE SHUTTLE
SPACE STATION
LAUNCH DATE:
JANUARY 1976
JULY 1978
JANUARY 1981
JULY 1983

Figure 1.

used for Skylab I. Each space station would be a unique craft having accommodations for a different sized crew and for a particular experiment program.

The SIV-B space station would be launched unmanned and it would be manned and logistically supplied by separate launches of Apollo command and service modules (CSM). These would rendezvous with the spacecraft and keep them manned continuously for the one-year durations of each mission. For this study the use of the current three-man version of the Apollo CSM was used as the basic supply and Earth return vehicle. A four-man version of the Apollo vehicle also has been considered as an alternative approach in this study, for it has been rather extensively studied, and it could be made available without extensive new vehicle development. Its modifications would consist of limited changes to its internal arrangements and to some systems. It has been assumed that each crew would remain in orbit an average of 90 days. The space station crew complements for each mission are shown in Figure 2. Mission A has a minimum crew size of three men but must house six men during logistic crew changes. Mission B has an average crew size of six men and mission C and D have nine men. The use of the four-man Apollo logistic vehicles are considered as alternates for only Missions C and D. The use of four-man Apollos reduces the number of logistic spacecraft and launches required by one-quarter, but it reduces the crew size by only one.

Three launch vehicles were considered to launch the crew and supplies to the orbiting station. They are each in a different state of technical development and availability. However, when one considers the large number of vehicles that are required, this development difference is of minor importance. The basic launch vehicle considered is the Saturn I-B. There are some of these available, but to complete this program, it would be necessary to reactivate its production. The second launch vehicle is the Titan III-M. This vehicle was essentially operational at the time the Air Force MOL program was cancelled; however other versions of this vehicle are currently in production and use. In using this vehicle, the integration of the Apollo CSM with this launch vehicle would be its primary development requirement. The third logistic launch vehicle considered is a new one. It would be developed by integrating the 120-inch seven-segment solid rocket

LOGISTIC FLIGHT SCHEDULE FOR INTERIM SPACE STATION MISSIONS

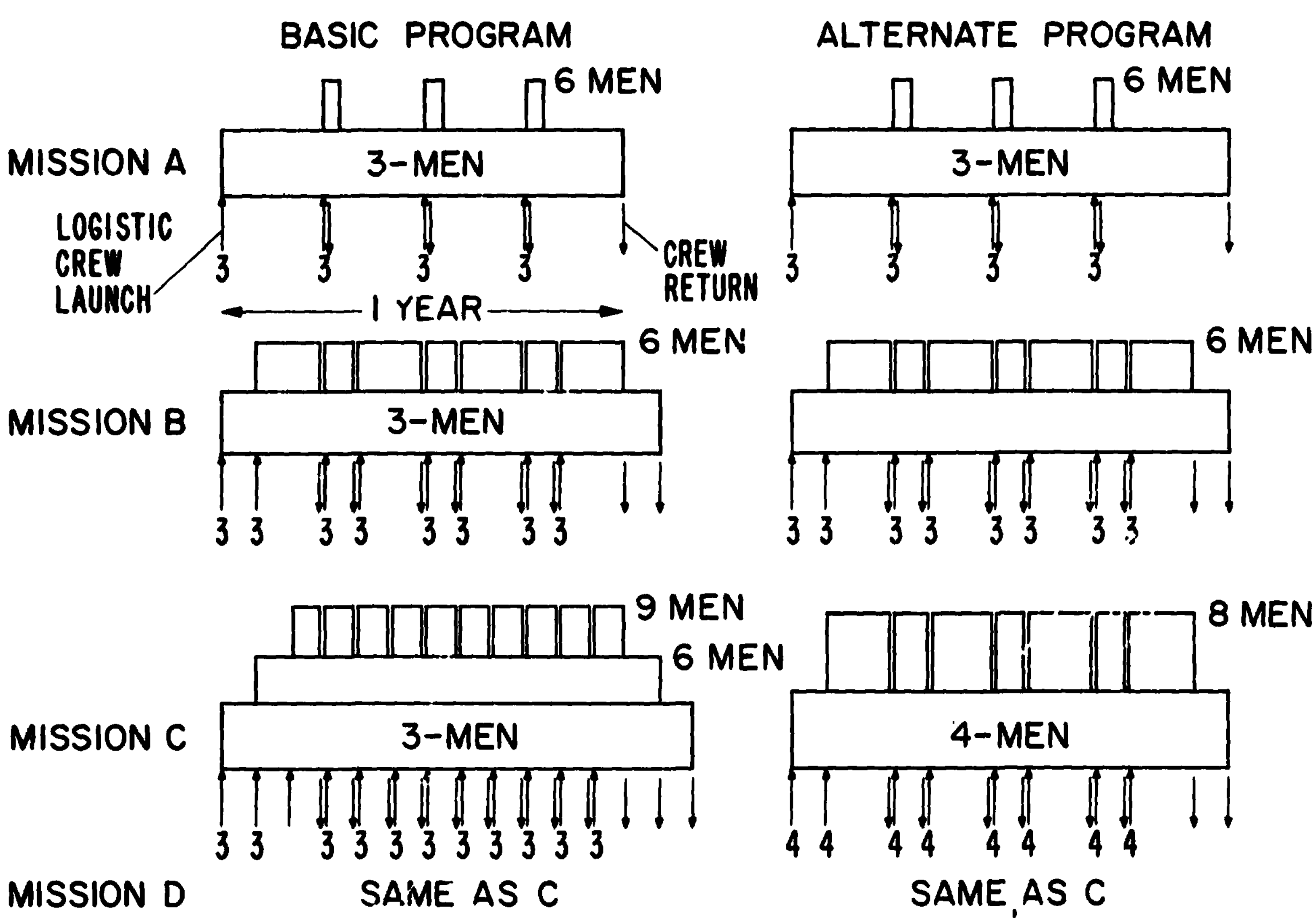

Figure 2.

motors of the Titan III-M with the Saturn IV-B stage. The major development item for this launch vehicle would be in the engineering, development, and operational testing of the integrated solid rocket motor first stage with the liquid rocket second stage.

Experiment Program

By nature, the experiment plan for this evolutionary space station program with the limited resources of spacecraft volume and manpower, is a compromise between technological capability and operational practicality. The evaluations and compromises that were necessary in the generation of the program were largely the result of subjective judgment. The rationale behind the experiment program recognized that the presently conceived NASA manned space flight plan has a significant "time gap" between the end of occupancy of Skylab I and the first manning of the Space Station. Skylab I is a proof type of mission in terms of operational concepts. It will supply a great deal of valuable information, but it will not be able to provide definitive answers for the many biomedical, scientific, and operational questions necessary to the design, and use of the Space Station. The crew size and capability is limited in terms of staytime (28 or 56 days), astronaut autonomy, inflight experiments, and facility maintenance and repair ability. The technology is largely that of the Gemini-Apollo systems and man's environment will be limited to zero "g". On the other hand, the Space Station will have advanced life support systems, large crews, variable gravity, and multi disciplinary experiments which are all advanced technologies in terms of spaceflight social-scientific knowledge. This study addresses its experiments to this technology gap and to those manned space experiments that would enhance the capability and usefulness of the Space Station if performed earlier.

What are the technology areas to which this evolutionary experiment plan should address itself? Perhaps first and foremost are those involved with man. There is needed information about how man lives, reacts, and functions in space for periods of at least 90 days. What are the changes in biological physical and mental states that take place; how effective is the crew member as an observer, experimenter, engineering technician, or para-medic; and could automated modes of operation or special devices enhance his capability? Another area involves the question of how to utilize and exploit the unique environment offered by a spacecraft in Earth orbit. There is a need to know what the constraints are that this environment places upon the experiment or function being performed, and how man may use his abilities to enhance or reduce the environmental effects. Another area of investigation is the effect upon experiments, perimental procedure, and the role of investigator as a flight experiment changes from a preprogrammed ground controlled status to full flight crew control and analysis status. Using the above criteria selection, the disciplines and experiments as proposed in NASA's document, "Candidate Experiment Program for Manned Space Stations", Ref. 1, were surveyed. This information, was supplemented by the results from North American Rockwell and McDonnell Douglas Companies' work during their Space Station studies, Refs. 2 and 3.

The rationale and constraints described above were used to define an experiment set for each mission. Another necessary factor considered was the amount of manpower that would be available and that would be required to perform each experiment. A feasible experiment workload for each mission was determined by assuming that 2/3 of the crew would be available for experiment work (the remainder would be concerned with normal operational and spacecraft maintenance tasks) and that the work week would consist of six days, each of ten hours duration. Further, an effectiveness of 75% was assumed, which resulted in a useful work week per man of 45 hours. The experiment sets that evolved out of this evaluation and synthesizing procedure are shown in Figure 3 for each of the four missions. Each mission has a major experiment discipline emphasis as well as several compatible minor disciplines. For ease in discussing the four missions, each will be designated by its primary emphasis as follows:

Mission A	Biotechnology
Mission B	Earth Resources and Applications
Mission C	Space Exploitation
Mission D	Astronomy

Scheduling and sequencing considerations for the various experiments are beyond the scope of this study and have not been examined. If they had been, in some cases it may have been impossible to perform all

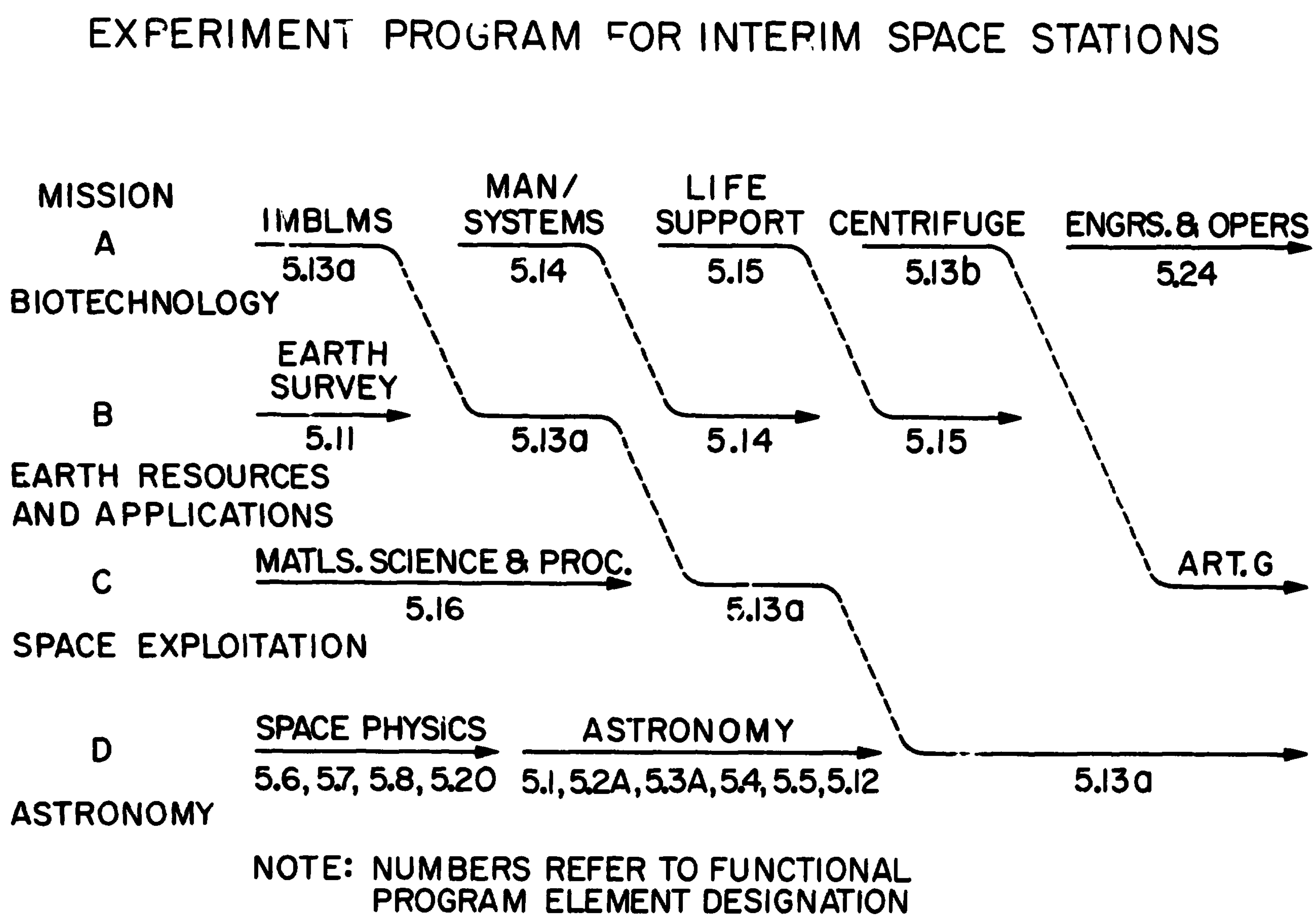
EXPERIMENT PROGRAM FOR INTERIM SPACE STATIONS
MISSION
A
BIOTECHNOLOGY
IMBLMS
5.13a
MAN/
SYSTEMS
5.14
LIFE
SUPPORT
5.15
CENTRIFUGE
5.13b
ENGRS. & OPERS
5.24
B
EARTH RESOURCES
AND APPLICATIONS
EARTH
SURVEY
5.11
5.13a
5.14
5.15
C
SPACE EXPLOITATION
MATLS. SCIENCE & PROC.
5.16
5.13a
ART. G
D
ASTRONOMY
SPACE PHYSICS
5.6, 5.7, 5.8, 5.20
ASTRONOMY
5.1, 5.2A, 5.3A, 5.4, 5.5, 5.12
5.13a
NOTE: NUMBERS REFER TO FUNCTIONAL
PROGRAM ELEMENT DESIGNATION

Figure 3.

experiments listed for each mission at the manpower levels specified. However, it is probable that serious conflicts generally will not occur because most of the selected experiments permit scheduling flexibility. It is also expected that the larger crew sizes, greater than three men, should permit greater flexibility and scheduling freedom, as well as requiring a smaller effective percentage of the crew time to maintain the spacecraft systems.

The aim of each mission and the resulting experiment set selected is discussed in the order of the missions. The characteristics of each experiment, namely, weight, power, and supporting requirements imposed upon the spacecraft are detailed. Some of the operational questions that each mission experiment program attempts to answer are indicated.

Biotechnology Mission - This mission, as planned, will continue the biomedical work started on the Skylab I mission and develop further an understanding of human factors and life support problems in these longer duration missions. The experiments selected for this mission and their supporting requirements are listed in Table 1. These experiments should help provide an understanding of man's ability to survive and function in space, since they would measure biomedical and physiological effects on each crew member of long-term exposure to zero gravity environment. An onboard artificial gravity simulator would be operated to test and evaluate its effects both upon man's condition and upon the spacecraft systems. Crew cycle and operational problems would be examined to evaluate the capability and reliability of the crew to maintain and repair the onboard systems under continuous zero gravity usage.

Some of the experimental equipment listed in Table 1, like the Integrated Medical and Behavioral Laboratory Measurement System (IMBLMS) device, would need to be built into the spacecraft systems. However, many related experiments might be supplied as carry-on items during later logistic supply flights. Many of these experiments would be associated with human factor and human capability investigations where it may be desirable to change or to upgrade the experiments. The peak power requirements for all experiments is about 3 1/2 kilowatts (KW) with an average requirement of 2 KW. This experiment electrical load is similar to that required for Skylab I. More detailed characteristics of each experiment

Table 1
Experiment Requirements for Mission A

EXPERIMENT CHARACTERISTICS

Experiment	Weight lbs.	Man-Hours Per Week	Average Power w	Peak Power w	Energy/Week w - Hr.	Maximum Data Rate bps	Data/Week bits
5:13 Biomedical and Behavioral							
IMBLMS	1300	45	500	1000	2.25×10^4	1.4×10^7	1.7×10^{11} (b)
Centrifuge	1720	55	250	250	1.37×10^4	2.5×10^4	9×10^8
5:14 Man-System Integration	500	12	750	1750	9×10^3	1.4×10^7	6.5×10^{10} (b)
5:15 Life Support and Protective Systems	(a)	6	See:	Note (a)			
5:24 Operational and Engineering	1000	12	300	300	3.6×10^3	Nominal	Nominal
TOTALS	4520	130	1800	3300			

Notes: (a) Requirements are included in the life support system weight, power, and housekeeping data.
(b) Based on a ten percent duty cycle for high resolution TV.

CONSUMABLES, LBS. FOR EACH 180 DAYS

Experiment	Operational			Maintenance		Spares		
	Initial	Resupply	Return	Initial	Resupply	Initial	Resupply	Return
5:13	120	120	120	20	15	60	15	10
5:14	125	125	125	5	2	30	10	10
5:15	60	60	30	7	2	30	10	8
5:24	2838	2838	156	25	7	724	350	235

for this mission as well as for the other three missions are given in Appendix A.

Earth Resources and Applications Mission - This mission is concerned primarily with the test and evaluation of earth resources sensor and data handling techniques. But in addition, the biomedical work, and the crew and protective system studies are continued. This mission assumes that man can perform useful and purposeful functions, and thus man is used extensively to operate and modify space borne equipment.

The earth survey investigations outlined for this mission are of an experimental rather than an operational nature. The five major disciplinary areas that would be studies are: 1) Agriculture/Forestry/Geography, 2) Geology/Minerology, 3) Hydrology/Water Resources, 4) Oceanography, and 5) Meteorology. The sensor sets planned for use in each area are listed in Table 2. Since the sensors are mainly being evaluated for their capability, the crew members are heavily involved in setting up, calibrating, analyzing results and modifying the sensors and instruments. Truth sites will be used extensively to provide data which can be correlated with the signatures of the sites as measured by the sensors. The crew members will be involved in onboard data processing, analysis, and interpretation; operation, maintenance, and modification of complex instruments; and the capability to respond to unprogrammed events and sighting opportunities. This mission will continue the strong medical and physiological crew evaluation program and it will stress the analysis of social interactions among crew members because of the larger crew.

The experiments selected for this mission and the support required from the spacecraft are listed in Table 3. The maximum data to be transmitted to Earth from the sensors puts a high short-time requirement upon the communication system; however, on a weekly average basis its requirement is about one half that necessary for the crew's medical, physical and mental status data.

Space Exploitation Mission - The prime objective of the third mission is to evaluate in terms of direct Earth economic benefits the use of the space environment for materials processing and manufacture. The continuing biomedical effort of the earlier missions will be expanded to include

Table 2
Sensor Set Summary

Sensor	Weight lb.	Volume ft^3	Average Power watt	Data Rate (bps) Science	Engin.
1. Metric Camera	360	35.0	504	60 (a)	17.6
2. Multispectral Camera	185	10.0	700	48 (a)	28.8
3. Multispectral IR Scanner	150	2.7	60	3×10^7	640
4. IR Interferometer Spectrometer	65	1.3	25	4×10^3	640
5. IR Atmospheric Sounder	45	2.3	85	1.7×10^4	162
6. IR Spectrometer/Radiometer	65	4.0	50	3.8×10^4	200 (c)
7. MW Scanner	76	26.0	25	10^2	1 (c)
8. Multifrequency MW Radiometer	50	67.0	150	480	16
9. MW Atmospheric Sounder	80	1.4	180	100	0.8
10. Radar Imager	620	53.0	1500	60 (a)	640 (c)
11. Active-Passive MW Radiometer	100	2.0	50	3.2×10^3	7.2
12. Visible Wavelength Polarimeter	50	5.1	10	160	16
13. UHF Sferics	22	1.3	6	260	1
14. Absorption Spectrometer	95	12.0	22	400	0.8
15. Laser Altimeter	371	12.0	636	24 (b)	8
16. UV Imager/Spectrometer	150	3.3	45	8×10^4	4
17. Radar Altimeter/Scatterometer	75	1.0	130	3.2×10^3	12.6
18. Photo-Imaging Camera	145	10.0	129	3.5×10^7	640 (c)
19. Data Collection	11	0.2	8	-	-

Discipline	Sensor Set
Agriculture/Forest/Geography	1, 2, 6, 7, 10, 18
Geology/Minerology	1, 2, 3, 6, 8, 10, 15, 16, 17, 18
Hydrology/Water Resources	1, 2, 3, 7, 8, 10, 18
Meteorology	4, 5, 8, 9, 12, 13, 14
Oceanography	1, 6, 8, 17

Notes:

(a) Pounds of film

(b) Per metric camera frame

(c) Estimated

Table 3
Experiment Requirements for Mission B

EXPERIMENT CHARACTERISTICS

Experiment	Weight lbs.	Man-Hours Per Week	Average Power w	Peak Power w	Energy/Week w - Hr.	Maximum Data Rate bps	Data/Week bits
5:11 Earth Surveys	4700	90	885	3480	8130	6.5×10^7	1.6×10^{11}
5:13 IMBLMS Only	1300	70	500	1000	4.5×10^4	1.4×10^7	3.4×10^{11} (b)
5:14 Man-System Integration	425	60	165	555	2×10^4	1.4×10^7	3.2×10^{11} (b)
5:15 Life Support and Protective Systems	2355	50	3300	6040	1×10^5	See: Note (a)	
TOTALS	8780	270	4850	11075			

Notes: (a) No data load.
(b) Based on a ten percent duty cycle for high resolution TV.

CONSUMABLES, LBS. FOR EACH 180 DAYS

Experiment	Operational			Maintenance		Spares		
	Initial	Resupply	Return	Initial	Resupply	Initial	Resupply	Return
5:11	1765	1765	1640	25	15	163	26	15
5:13	120	120	120	20	15	60	15	10
5:14	125	125	125	5	2	30	10	10
5:15	60	60	30	7	2	30	10	8

the rotation of the entire spacecraft about an axis between the mission module and a suitable counterweight such as the Saturn II launch stage. This arrangement should be sufficient to minimize rotational coriolis effects and to closely simulate, for evaluative purposes, the artificial gravity environment as proposed for the Space Base.

The materials processing experiments are selected to establish the feasibility of processing and manufacturing products which require the near zero-gravity or extremely clean high vacuum environment of space. Various metal compositing, thin film, crystal growing, and biological and chemical compounding tests have been selected to make up this experiment package. Only methods whose products have a potential value greater than that added by the cost of space transportation have been considered.

The experiments for this mission and their supporting requirements are listed in Table 4. The artificial gravity experiments would be performed early in the mission so that the Saturn II stage counterweight could be abandoned as soon as practical, Ref. 4. It should be noted that the experimental equipment is the same for either an eight-man or a nine-man mission. The change of crew size is reflected mainly in the logistic support and data return loads. These large crew sizes will limit the living and work room each man has at his disposal, and some interesting results should come out of this mission as to social and human tolerance levels.

<u>Astronomy Mission</u> - The fourth mission places its major emphasis upon astronomy, capitalizing upon the experience gained in the solar telescope experiments of Skylab I. This orbiting manned module would assume the role of a basic scientific research laboratory. It would also examine the experimental and operational aspects of attached, tethered, or free flying experiment modules, for the preferred accommodation for the stellar and the solar astronomy experiments is the free flying mode Spacecraft stability, control, and positioning and the spacecraft outer environment must all meet demanding standards. Although by this fourth mission a fair amount of data should have been accumulated regarding human capability in and reaction to the space environment, the medical and physiological testing of the crew members would be continued. The IMBLMS would be in its fourth

Table 4
Experiment Requirements for Mission C

EXPERIMENT CHARACTERISTICS

	Experiment	Weight lbs.	Man-Hours Per Week	Average Power w	Peak Power w	Energy/Week w - Hr.	Maximum Data Rate bps	Data/Week bits
5:16	Materials, Science and Processing (a)	1200	95	2000	5000	1.3×10^4	-	2.1×10^8
-	Artificial Gravity (b)							
	Space Station	12600	-	5000	8000	1.5×10^5	-	-
	Launch Vehicle	7900	-	-	-	-	-	-
5:13	IMBLMS Only (c)	1300	130	500	1000	8.4×10^4	1.4×10^7	6.3×10^{11}
5:14	Man-System Integration (d)	425	70	165	555	2.4×10^4	1.4×10^7	3.8×10^{11}
TOTALS		23425	295	7665	13555			

Notes: (a) Operates only under zero gravity conditions.
(b) Performed during a 45-day period.
(c) Operates during entire mission.
(d) Is performed only in conjunction with artificial "g" experiment.
(e) Propellants.

CONSUMABLES, LBS. FOR EACH 180 DAYS

Experiment	Operational			Maintenance		Spares		
	Initial	Resupply	Return	Initial	Resupply	Initial	Resupply	Return
5:16	382	382	247	10	5	40	20	15
Artificial "g"								
Spacecraft	8100 (e)	-	-	-	-	-	-	-
Launch Vehicle	13900 (e)	-	-	-	-	-	-	-
5:13	120	120	120	20	15	60	15	10
5:14	125	125	125	5	2	30	10	10

revision, and the amount of time required by the examiner and the subject should be approaching a minimum value to secure the desired data.

The experiments and their supporting requirements are listed in Table 5. These experiments require a fair amount of electrical energy which would be reflected in additional capability and weight of the electrical power system. Weight has been included in the stellar and solar astronomy experiments to account for their module support.

Experiment Orbit Constraints - In reviewing [illegible] of the experiments for the four missions so as to determine their [illegible] desirable or advantageous Earth orbital altitude and inclination, it [illegible] apparent that most of the experiments placed few operational constraints on the spacecraft. These few conditions are summarized in Table 6. Except for the Earth Resources and Application Mission, it is apparent that some other mission or system requirements will be the determiners to establish orbits. This mission should overfly the most populace and agriculturally productive areas of the Earth, and so its inclination should be 50 degrees.

Scientific Accomplishments - One other evaluation was made of this experiment program. That is, what impact will this program have upon the program suggested for the Space Station? Figure 4 shows the accumulated rate of accomplishment of experiments for each of the four missions in terms of man-hours expended upon each of the experiments. These accomplishments are compared with the expected rate possible on the Space Station. As can be seen, the four evolutionary space station missions accomplish more than two Space Station years' equivalent experiments. There is a more important item that cannot be measured analytically, and that is the flight experience, technology and operational development, and the human competence that would be gained through this interim program. If this program were completed, it would enable the Space Station to start its effective experimental usefulness almost at initial manning, for most of the human and operational uncertainties of long duration spaceflight would have been removed by the results of these four earlier interim space station flights.

SPACECRAFT SYSTEMS

If as much as possible of previously developed and operationally

Table 5
Experiment Requirements for Mission D

EXPERIMENT CHARACTERISTICS

Experiment	Weight lbs.	Man-Hours Per Week	Average Power w	Peak Power w	Energy/Week w - Hr.	Maximum Data Rate bps	Data/Week bits
5:2 Stellar Astronomy (a)	6000	40	850	850	3140	3.2×10^6	1×10^{11}
5:3 Solar Astronomy (a)	6195	25	595	595	5950	-	-
5:4 UV Stellar Survey	1300	15	120	120	710	-	-
5:6 Space Physics Airlock Experiments	95	5	105	235	105	-	-
5:8 Cosmic Ray Physics Laboratory (a)	26700	20	7500	7500	6.5×10^4	1×10^5	5.2×10^7
5:13 IMBLMS Only	1300	130	500	1000	8.4×10^4	1.4×10^7	6.3×10^{11}
TOTALS	41590	235	9670	10300			

Notes: (a) Separate module desirable.
(b) Module required.

CONSUMABLES, LBS. FOR EACH 180 DAYS

Experiment	Operational			Maintenance		Spares		
	Initial	Resupply	Return	Initial	Resupply	Initial	Resupply	Return
5:2	200	200	200	5	2	50	20	20
5:3	405	405	405	35	10	112	33	33
5:4	26	26	26	5	-	11	3	2
5:6	45	45	45	2	2	20	5	3
5:8	320	1120	560	29	12	140	25	25
5:13	120	120	120	20	15	60	15	10

Table 6
Experiment Ortital Constraints

Mission	Altitude	Inclination
A - Biotechnology	Not Critical	Not Critical
B - Earth Resources and Technology	< 250 Nautical Miles	Minimum Useful: 25° Preferred: 45° Selected: 50°
C - S[illegible]ce Exploitation	≥ 200 Nautical Miles	Not Critical
D - Astronomy	≥ 200 Nautical Miles	Not Critical

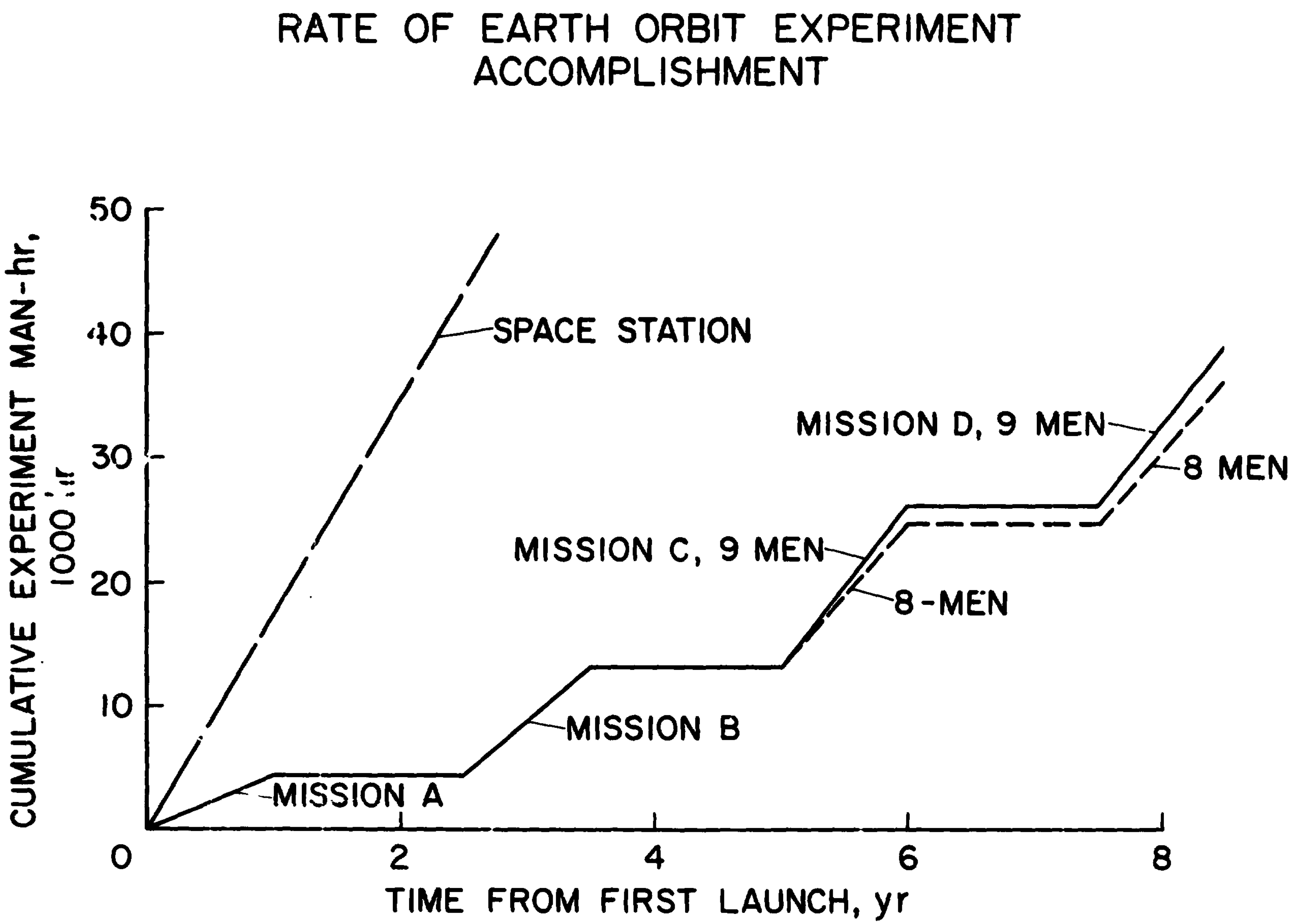
RATE OF EARTH ORBIT EXPERIMENT
ACCOMPLISHMENT
CUMULATIVE EXPERIMENT MAN-hr, 1000's
50
40
30
20
10
0
2
4
6
8
TIME FROM FIRST LAUNCH, yr
SPACE STATION
MISSION A
MISSION B
MISSION C, 9 MEN
8-MEN
MISSION D, 9 MEN
8 MEN

Figure 4.

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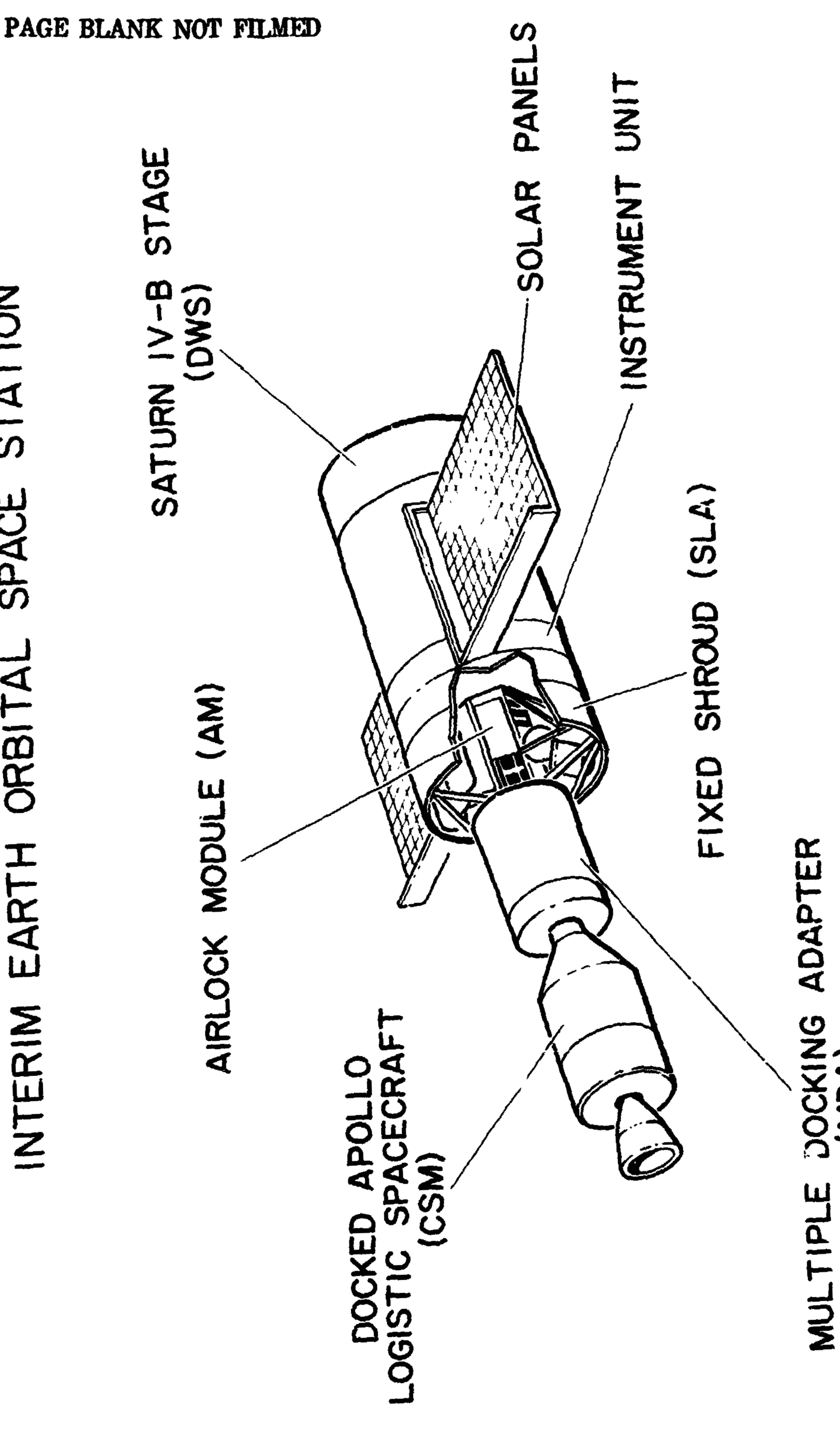


Figure 5.

Table 7
Dry Saturn IV-B Stage Characteristics

Structure

Saturn IV stage, 22 feet diameter, 10,800 cu. ft. vol.

Meteoroid shield deployable

Solar array attachment and deployment mechanism

Internal insulation

IVA and EVA attachments and hand holds

Living and laboratory compartments

Removal of all J-2 engine hardware

Use of unused oxygen propulsion tank for waste disposal and storage

Crew Systems

Compartment for each crew member, includes personal effects and sleep station

Pressure suits for EVA and IVA

Non-pressurized garments

Wardroom to permit astronaut recreation, relaxation, conditioning, and eating

Compartment for crew medical monitoring

Astronaut mobility and maneuvering aids

Freeze-dried, frozen and fresh food stored

Waste management contains fecal and urine collection and oral and body cleansing facilities and collections

Environmental Control

Supplies for 3 months of operation plus 1 month emergency supply

Two gas atmosphere at 5 p.s.i.a. total pressure

Oxygen partial pressure minimum of 3.7 p.s.i.a.

Nitrogen partial pressure of 1.3 p.s.i.a.

Relative humidity of 50% nominal

Carbon dioxide and contaminant control by means of regenerable molecular sieves and activated charcoal filters

Temperature maintained at 70° F

Water supply stored for consumption and cleanliness with hot and cold available

Table 7, continued
Dry Saturn IV-B Stage Characteristics

Electrical Power

Solar cell array generated power

Transfer of power to or from other modules

Rechargeable Nickel Cadmium batteries

Two wire distribution of D.C. power

A.C. Inverters at equipment needing alternating current power

Communications

Two-way voice via MSF Network

Two-way television via MSF Network

Transmit television, real and delayed-time data, subsystem status, biomedical monitoring and experiment support in down-link

Teleprinter up and down link

Television intercommunications between components and modules

Antennas with or without CSM docked

Attitude Control

Two module thrusters with outputs from 10 to 100 pounds

Three control moment gyros - one for each axis

CMG desaturation by the thruster modules

Thruster fuel is bi-propellant

Two digital computers and two analog pointing assemblies

Control commands from ground or AM

Sun and horizon sensors

Inertial stable platform and signal generator of this unit. The characteristics of the airlock module are given in Table 8. The multiple docking adapter furnishes the interface with the Apollo logistic CSM having docking ports to which they dock. The in-orbit support for the logistic craft during storage is supplied from the orbiting spacecraft through this component. Table 9 lists the characteristics of this module. The Skylab I documents, Refs. 6, 7, 8 and 9, give more detailed specifications for these components.

Other Saturn and Skylab originated components that would be used for launch operations are the instrument unit (IU) and payload shroud. The instrument unit furnishes the guidance and operational control for the launch vehicle and spacecraft from liftoff through orbit insertion and until manned. Its functional characteristics are given in Table 10. The payload shroud or nose cone would be the same as that used to protect the Skylab MDA and AM components during launch, and it would protect similar components on the interim stations. The shroud can be jettisoned either at propulsion staging or after orbit insertion.

Electrical Power

This section discusses the rationale of power system selection for the four interim space stations and a possible power system evolutionary development to satisfy the needs of this program as well as those of the larger Space Stations and Space Bases. The factors which were considered in the selection of the power systems are: power requirements, operating environment and performance requirements, power system availability and performance, and special factors which may influence system selection.

Power Requirements - The estimate of the range of power requirements for each of the interim space station missions is given in Figure 6. The top of the circle indicates the approximate peak power demand, and the bottom indicates about the minimum needed to sustain crew and spacecraft systems. Current plans for Skylab I place its average total requirements at about 6 kilowatts to support all systems. In Appendix B details are given of the Skylab I power requirements, and the estimated power needs for several studied Earth orbital spacecraft. The range of electrical power requirement for each of the interim space station missions was constructed by using this Skylab I data and adding to it the needs of greater crew size

Table 8
Airlock Module

Structure

Transition between workshop and multiple docking adapter

Pressure tight airlock compartment

Support for environment gas storage

Environmental Control

Control temperature and humidity

Mount for heat rejection radiator for spacecraft

Regulate O_2 for EVA or IVA

Control atmosphere to 5 p.s.i.a. total pressure

Control oxygen at 3.7 p.s.i.a.

Control nitrogen at 1.3 p.s.i.a.

Carbon dioxide and contaminant control by means of regenerable molecular sieves and activated charcoal filters

Operation independent from other spacecraft modules

Electrical Power

Distribution center for spacecraft

All power conditioning units

Back-up batteries

Crew Provisions

Connections for IVA or EVA support umbilicals

Visual and audible display for spacecraft systems and safety

Communications

Interconnection for voice with other spacecraft modules

Outlet for monitoring television

Spacecraft systems monitoring to MSF network

Table 9
Multiple Docking Adapter

Structure

Ports for docking of logistic vehicles - Apollo

Pressurized to standard workshop conditions

Viewing ports

Environmental Controls

Temperature and humidity control by heat exchange

Coolant lines for experiment support

Atmosphere circulated from airlock module

Electrical Power

Supplied from airlock module distribution system

Interconnection to docked vehicles

Attitude Control

Primary control and display for thruster system

Back-up control of spacecraft

Data Management

Pick-up for television monitor

Table 10
Instrument Unit

Structure

Interface and support between dry workshop and shroud

Control System

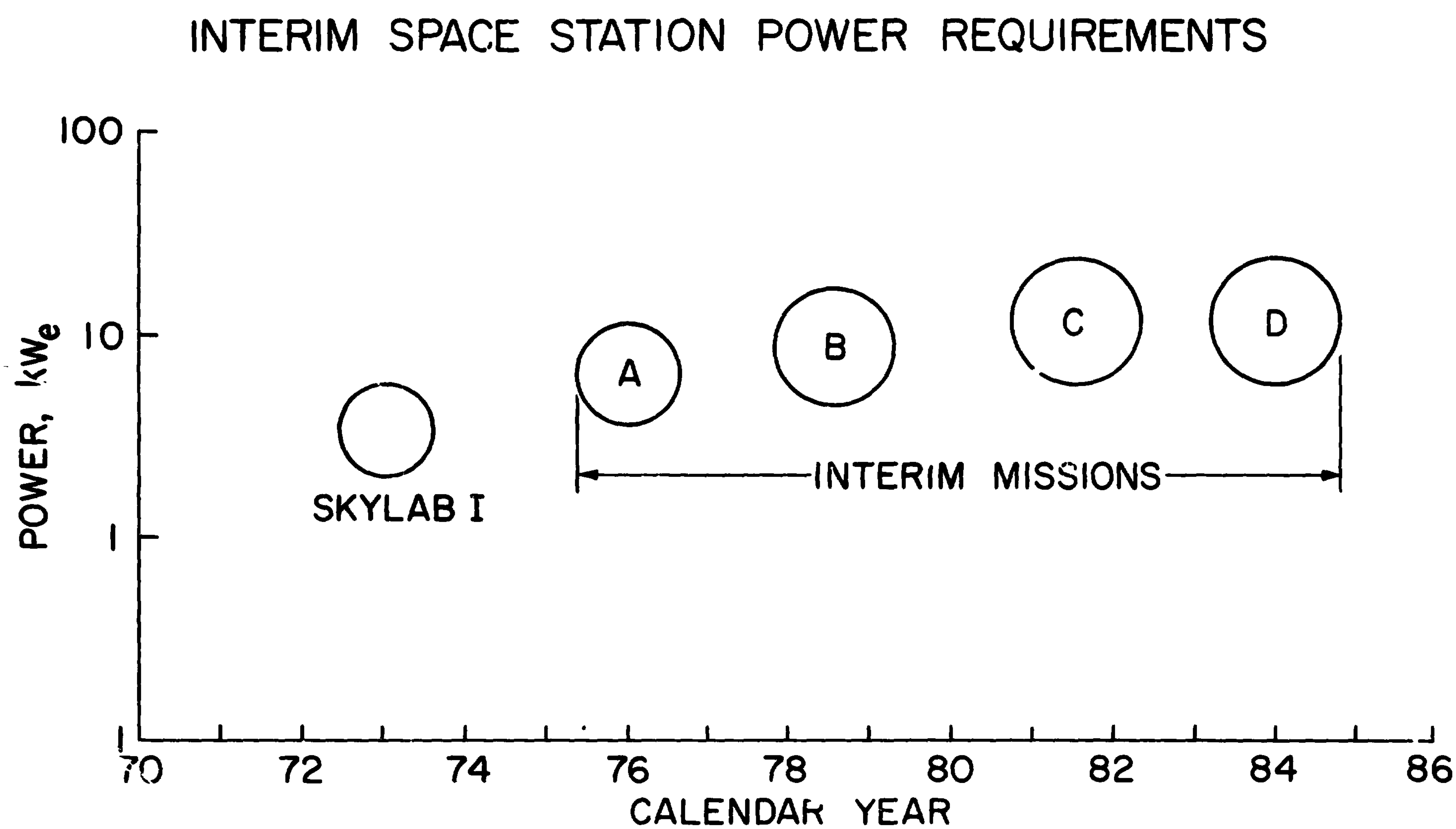
Sequences flight functions including placing spacecraft into orbit

Command link between spacecraft and ground during early flight phases

Gives guidance, navigation and attitude functions to spacecraft

Electrical Power

Non-rechargeable batteries



INTERIM SPACE STATION POWER REQUIREMENTS
100
10
1
POWER, kW_e
SKYLAB I
A
B
C
D
INTERIM MISSIONS
70
72
74
76
78
80
82
84
86
CALENDAR YEAR

Figure 6.

support, changes in technology sophistication of systems, such as the life support system, and unique needs imposed by the experiment program.

Power System Selection - Those systems which have the potential for satisfying the interim space station power requirements are solar cells, fuel cells, Isotope Brayton, reactor Brayton, reactor thermoelectric, and reactor mercury Rankine. Each of these six power system types could satisfy the interim space station requirements. Only solar cells and fuel cells are technologically ready at the present, and it would be at least 1975 before the others are available. The first interim station power system should not be expected to deviate greatly technically or physically from the Skylab I spacecraft which is shown in Figure 7. If power requirements increase, solar arrays can be added in several different ways such as is shown in Figure 8. As can be seen, there should be no problem in accommodating solar arrays which would supply up to 20 KWe for any of the interim space stations. Solar arrays will therefore be the main source of power for each of the interim space stations.

If reactor or isotope systems are to be used to fulfill future space station power requirements, and such plans are indicated, it is logical to assume that the interim stations would be used to test such systems. These advanced systems would probably be launched separately as self contained modules and docked to the already orbiting space station. There are several reasons for the separate launch: 1) emergency abort procedures which ensure safety from nuclear effects are made simpler; 2) one launch configuration could be used for several space station configurations; and 3) the reactor systems are so heavy (between 30 and 130 thousand pounds) that it would require almost the full launch capability of a Saturn V derivative to launch it integrally with the mission station. The Titan class of launch vehicles might be capable of putting one of the early isotope systems into orbit. For nuclear systems with weights greater than about 20-30 thousand pounds, a larger launch vehicle such as the Intermediate 21 may have to be used to launch the system. Two ways in which suitable nuclear power modules may be configured with the space station are shown in Figure 9. In each module, the power conversion and radiator systems are located between the space station and the nuclear energy source and these supplement the shielding by adding some separation distance.

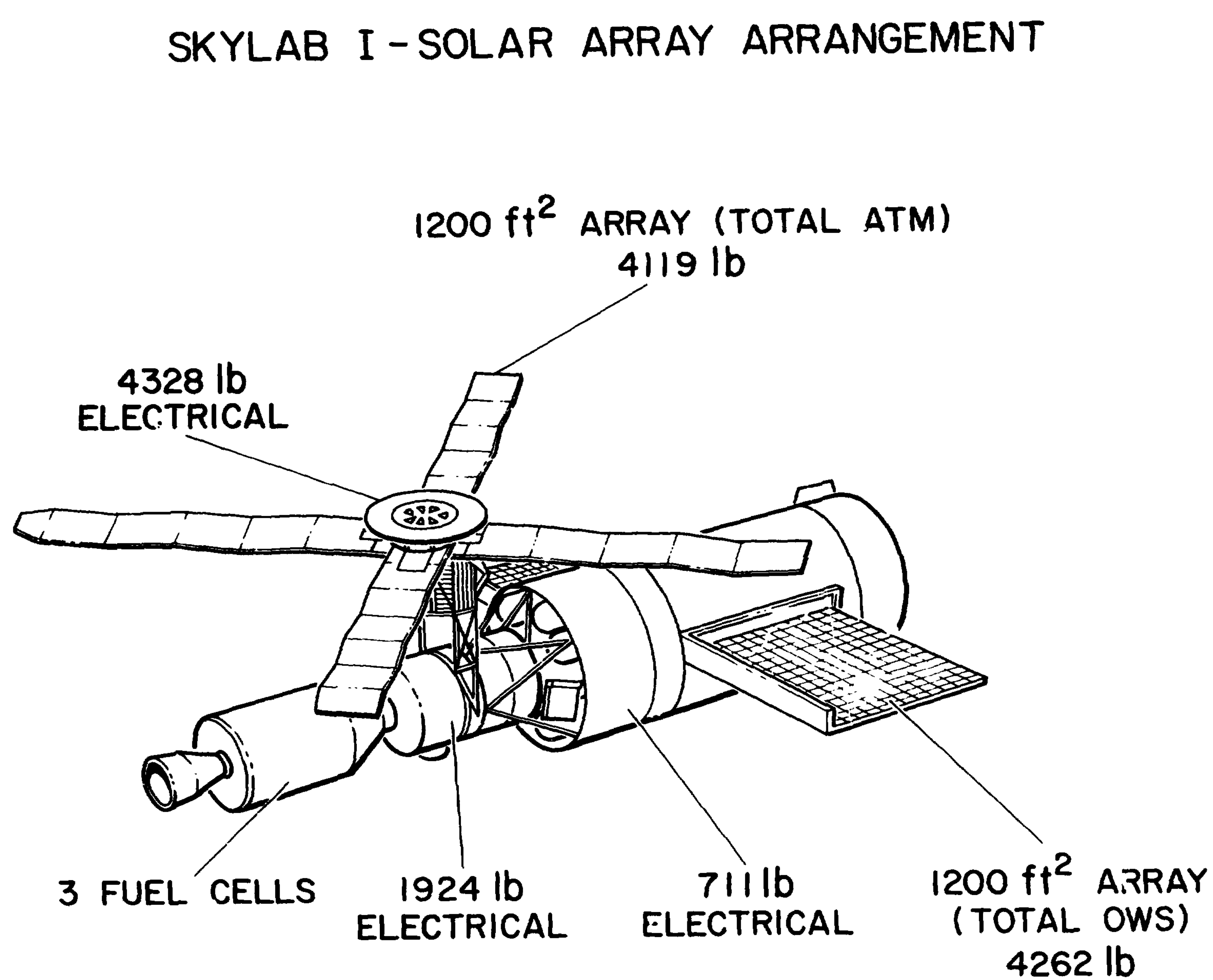

SKYLAB I - SOLAR ARRAY ARRANGEMENT
1200 ft² ARRAY (TOTAL ATM)
4119 lb
4328 lb
ELECTRICAL
3 FUEL CELLS
1924 lb
ELECTRICAL
711 lb
ELECTRICAL
1200 ft² ARRAY
(TOTAL OWS)
4262 lb

Figure 7.

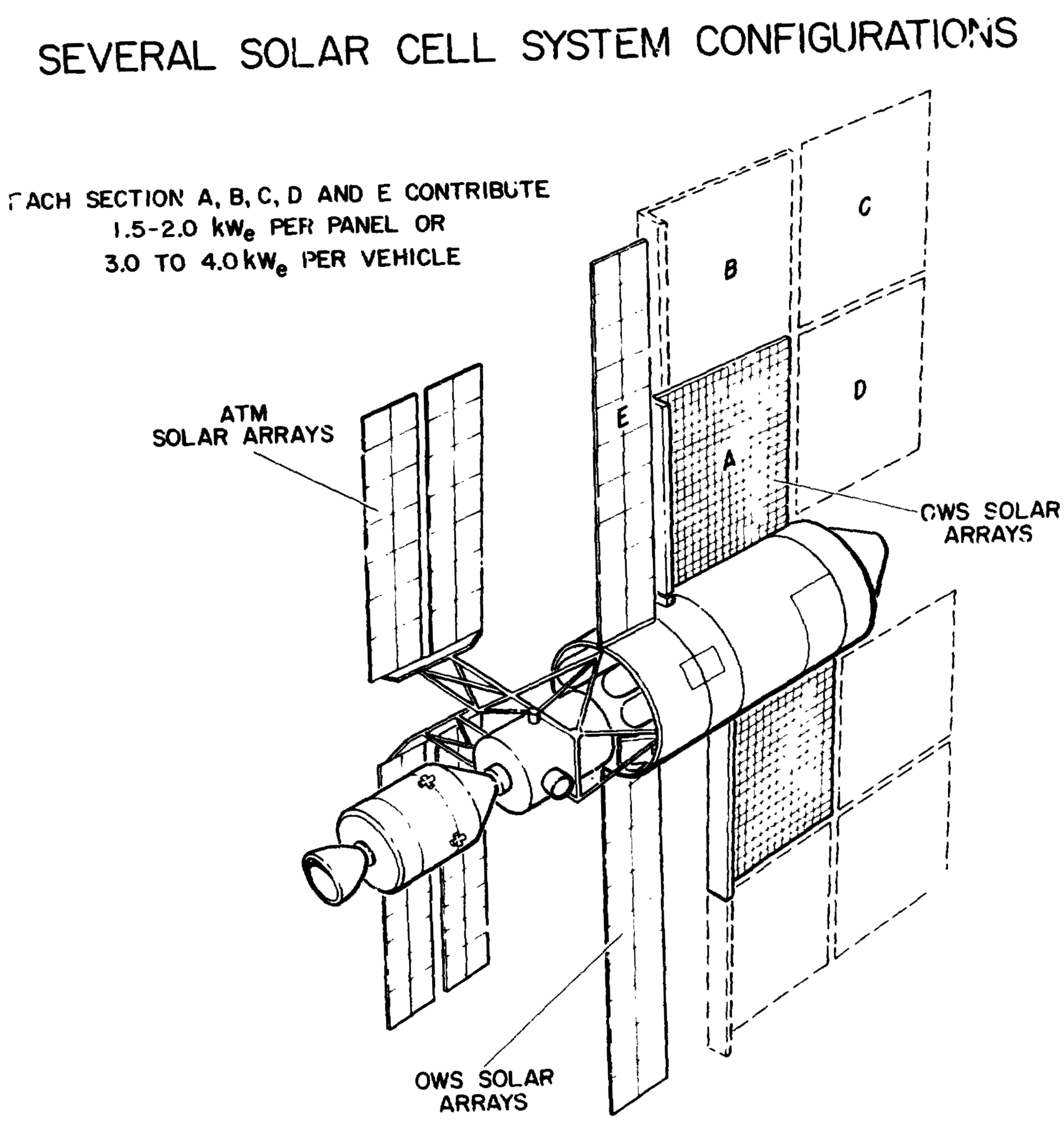

SEVERAL SOLAR CELL SYSTEM CONFIGURATIONS
EACH SECTION A, B, C, D AND E CONTRIBUTE
1.5-2.0 kW_e PER PANEL OR
3.0 TO 4.0 kW_e PER VEHICLE
ATM SOLAR ARRAYS
OWS SOLAR ARRAYS
OWS SOLAR ARRAYS
A
B
C
D
E

Figure 8.

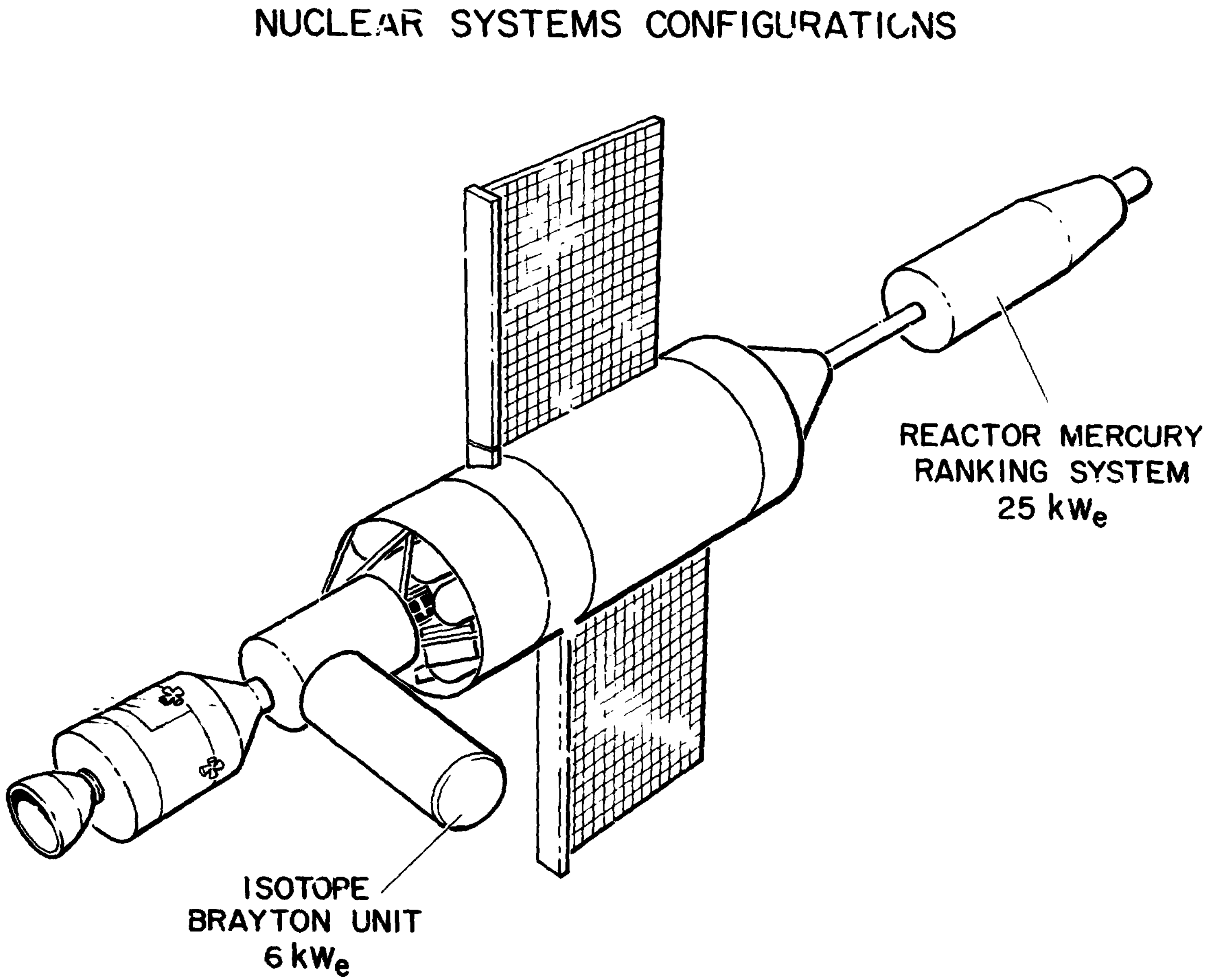

NUCLEAR SYSTEMS CONFIGURATIONS
REACTOR MERCURY
RANKING SYSTEM
25 kWe
ISOTOPE
BRAYTON UNIT
6 kWe

Figure 9.

Indications are that solar cells will continue to be used to satisfy the bulk of the spacecraft power requirements through the 1970's and even beyond. As power requirements grow from the 5-20 kilowatt level for the interim space stations to the 25-100 kilowatt level for the Space Station and Space Base, power system tradeoffs or usages between solar cells, isotope, or nuclear systems will depend upon the rate of technical development, actual power requirements, safety, and spacecraft configuration. The characteristics of the primary solar cell systems for each mission are given in Table 11. The solar cells are supplemented during orbital blackout and peak demands by nickel-cadmium batteries which can constitute up to half the total system weight indicated depending upon the orbit characteristics and the peak power requirements. The solar cell system total weight is shown to be the same for each mission. This result is due to expected advances in the technical development of solar cells and batteries during the time period for these missions. The isotope system is selected for the secondary power source, for it is most appropriate in terms of power level and expected technology development and also because it is the ty which is being given most serious study f the Space Station application.

Life Support System

The life support and protective system of the interim space stations uses as much of the Skylab developed hardware as might be appropriate. However, with four missions and an increasing number of crew members on board each subsequent spacecraft, it is necessary to augment these components and desirable to reclaim some of the waste products. A part of the experiment program of each mission are some tests which examine the operation of onboard systems and some which test non-space proved methods of crew support. In the earliest missions these experiments would consider first the methods for water recovery, then in later flights, oxygen recovery, and eventually the trial use of these recovery methods in flight type systems. This section will detail the characteristics of the life support systems on each of the mission spacecraft, and will discuss the serial technology development and changes in these systems between subsequent missions.

Table 12 shows the components that comprise each life support system.

Table 11
Interim Space Station Power System Characteristics

Mission	Average Continuous Power Required (KWe)	Primary System (Power)	Primary System Weight (lb)	Secondary OR Test System (Power)	Secondary OR Test System Weight (lb)
Skylab I	6	solar cells (/ KWe)	15,000	-	-
Space Station A	6-10	solar cells (6-10 KWe)	15,000	-	-
Space Station B	7-15	solar cells (7-15 KWe)	15,000	-	-
Space Station C	8-20	solar cells (7-15 KWe)	15,000	Isotope Brayton (6 KWe)	10,000
Space Station D	8-20	solar cells (7-15 KWe)	15,000	Isotope Brayton (10-15 KWe)	13,000

Table 12
Life Support System Components

Two-gas Atmosphere Control

Thermal Control

Humidity Control

Carbon Dioxide Removal

Hot and Cold Water Management

Food Storage and Preparation

Contaminant Control

Waste Management

Recreation and Sleeping Facilities

Compartmentalization

In this study not all of the components were considered in the same detail. Each component was evaluated by parametric analysis so that at least its effects upon system and spacecraft weight could be determined. The two areas of the life support system that effect the weight of the spacecraft and its logistic supply requirements the most are the atmosphere control and the water supply system. These system areas were studied in more depth and the results will be discussed in some detail.

Atmosphere Control - The atmosphere for each of the interim space stations will be the same. They will have a design pressure of 5 pounds per square inch absolute (psia). The primary gas will be oxygen and it will be maintained a. the necessary human level of 3.7 psia. The diluent gas will be nitrogen. Each one of the large number of necessary ports, throughputs and windows in the spacecraft are potential sources for gas leakage. Table 13 gives the assumed leakage of atmosphere gases from each spacecraft component, and these are the criteria for each of the interim space stations. The component leakage rates are the same as have been assumed for Skylab I. Based on this criteria, Table 14 gives the gas needs and defines the storage requirements. The leakage of the cabin atmosphere to space is a major item, and as such, it places a constraint upon the selection of cabin atmosphere pressure. For example, if the cabin atmosphere were increased from 5 psia to that normal for Earth, it would increase the leakage rate by a factor of three. The atmosphere gas resupply load is one of the major logistic items, and there is not a good biomedical reason to make the logistic load even greater. There are, however, several operational reasons which make the assumed design cabin pressure of 5 psia more desirable. One, is the desire to minimize the quantity of resupply gas which must be transferred between the logistic vehicle and the interim space station. Another reason is the desire to have the space suit and the cabin pressures about equal so as to alleviate the hazards of decompression sickness if accidental decompression should occur. It would be expected that technological and manufacturing improvements could reduce the rate of gas leakage, but in this study no assessment will be made for the effects of such expected lower leakage rates.

Because it is not easy to transfer or handle the resupply gases, any method which can more easily supply these gases would be attractive.

Table 13
Daily Atmospheric Gas Leakage Rates

Leakage (14 lb/day Total, 10.712 lb/day O_2, 3.288 lb/day N_2	
CSM	2.4 lb/day total
CSM/MDA Interface	1.2 lb/day total
One MDA Docking Port	0.2 lb/day total
MDA	1.8 lb/day total
AM/MDA Interface	0.6 lb/day total
AM	2.8 lb/day total
OWS	5.0 lb/day total
Molecular Sieve (3.3 lb/day Total, 1.42 lb/day O_2, 1.88 lb/day N_2)	
EVA Lock Repressurization (154 Cu Ft at 50° F with 7 Repressurization Cycles Each Week)	
EVA O_2 (9.0 lb/man hour)	
Leakage from O_2 and N_2 Supply Tanks and System Negligible	

Table 14
Gaseous Oxygen and Nitrogen Requirements

Oxygen	
Metabolic	2 pounds per man-day
Leakage (all)	10.75 pounds per day
Molecular Sieve	1.4 pounds per day
AM & MDA Repressure	40 pounds/each*
OWS Repressure	300 pounds/each*
EVA Airlock	1 pound/each use
EVA	9 pounds/man hour
Emergency & Contingency	10% total
Nitrogen	
Leakage (all)	3.35 pounds per day
Molecular Sieve	1.9 pounds per day
AM & MDA Repressure	12 pounds/each*
OWS	95 pounds/each*
EVA Airlock	.35 pound/each
Contingency	10% total

Note: * One repressurization each 90 days

One method is to supply oxygen in its combined form as water. Water can be easily containerized, and the spacecraft would require only a water electrolysis unit to free the oxygen. Figure 10 shows the total oxygen requirement for the four basic missions and for the missions with the optional crew size. The weight shown is for the gaseous oxygen and its tankage, Ref. 10. For safety and emergency requirements, a 120-day supply of gaseous oxygen should be the minimum amount on the spacecraft at launch. This supply, in terms of space station weight at launch, is between 15,000 pounds for Mission A and 21,000 pounds for Mission D. If water is used as the source for spacecraft oxygen, there would be between a three and a four-fold saving in just the tankage. Its disadvantages are the weight and maintenance required of the electrolysis cells, and the added capacity from the power source to supply the necessary electricity, Ref. 11. The implications of only a water electrolysis system for these missions were not considered and are not shown in Figure 10. Rather, it should be and is considered in conjunction with oxygen recovery.

The current technological development of the oxygen recovery and water electrolysis methods have just about kept pace with each other. Of several oxygen recovery methods being developed, the Sabatier unit is the simplest, and it has had the greatest flight type test experience. Oxygen recovery and water electrolysis are importantly interrelated functionally in that the Sabatier unit converts carbon dioxide to water and acetylene, and the water electrolysis cell separates the water to oxygen and hydrogen. The hydrogen is recycled to the Sabatier unit as it is required in its process and the oxygen is recycled to the cabin atmosphere. In the lower portion of Figure 10, there is shown for the four missions, the fixed equipment weights for the Sabatier carbon dioxide reduction and water electrolysis system. The expendable water and container weight for each of the four missions also are indicated. The net savings for each mission is not a simple subtraction of the weight below the abscissa from that above, for no consideration is included for the weight required by the electrical power system to supply the needs of the oxygen recovery system. However, the net weight savings that might be expected from the recovery of oxygen for these particular missions could be between 12,000 and 18,000 pounds.

In designating the types of atmosphere control system for each mission,

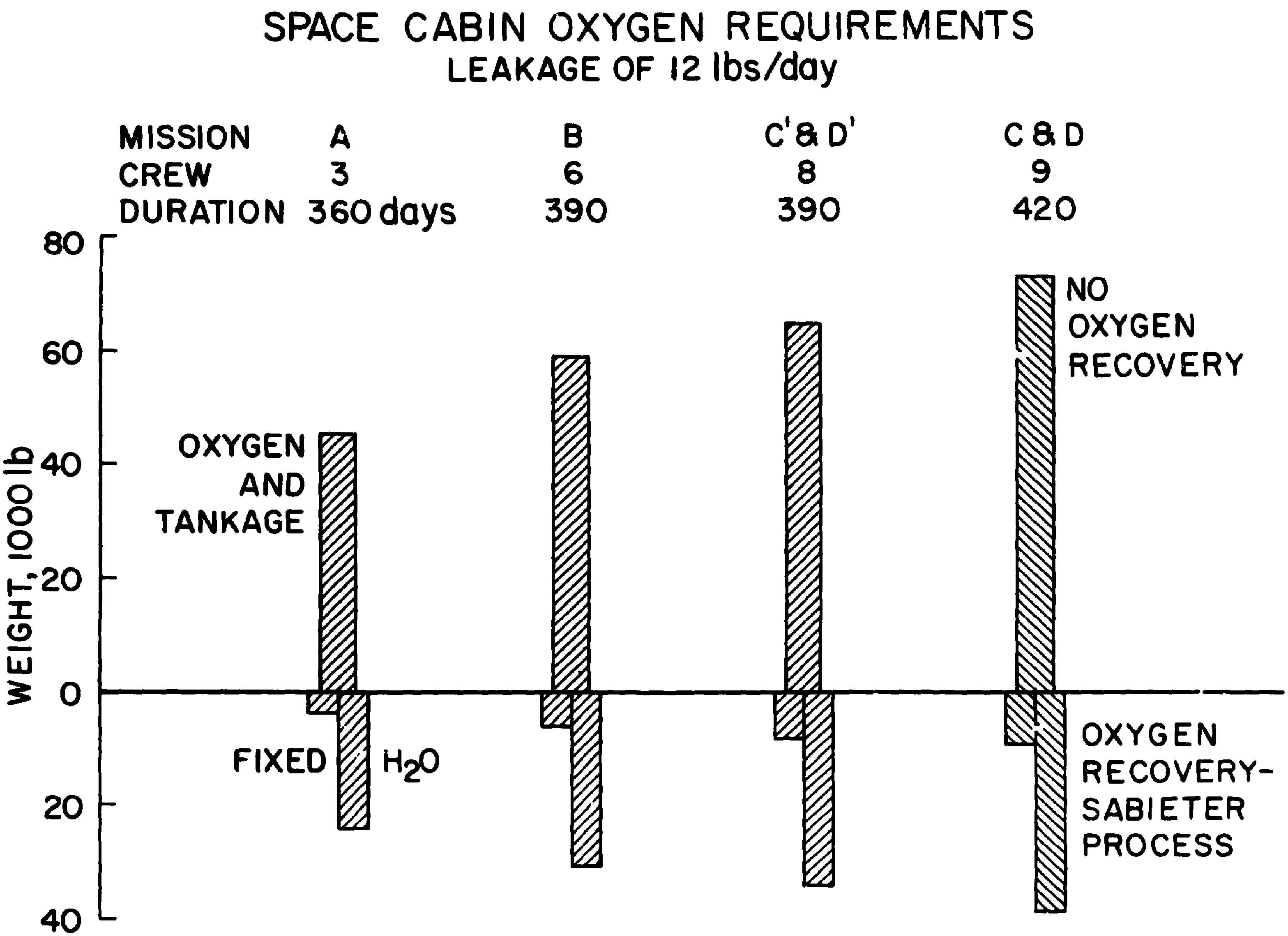

SPACE CABIN OXYGEN REQUIREMENTS
LEAKAGE OF 12 lbs/day
MISSION A B C'& D' C & D
CREW 3 6 8 9
DURATION 360 days 390 390 420
WEIGHT, 1000 lb
80
60
40
20
0
20
40
OXYGEN AND TANKAGE
FIXED
H_2O
NO OXYGEN RECOVERY
OXYGEN RECOVERY-SABIETER PROCESS

Figure 10.

it was assumed that the atmosphere system would consist of a stored gas system for Missions A and B. However, Mission B would have as part of its experimental payload an electrolysis unit capable of satisfying the spacecraft oxygen requirements. During its operation and trials, it would furnish the needed oxygen and thus it could save gaseous oxygen and in turn reduce the logistic oxygen loads. Mission C would have the oxygen supplied by water electrolysis units qualified during Mission B, and pressurized oxygen gas would be carried only for the cabin repressurization or emergency needs. During Mission C, a Sabatier carbon dioxide recovery system would be an experimental item, and it would be flight qualified on this mission. The atmosphere control system for Mission D would include the Sabatier and the water electrolysis units as a part of its spacecraft system. For Mission D, there would be almost a 20,000 pound weight saving in its atmosphere control system over that of an equivalent gas storage system. Lesser savings might be possible for the earlier mission; however, they are dependent upon the successful operation of the recovery systems during qualification trials. This advantage should not be considered in design, but instead should be taken as an advantage to be realized only during logistic supply of an actual mission. The weights for the atmosphere control systems are summarized later in this report both in terms of the entire spacecraft and of its logistic supply requirements.

Water Management - Both Skylab I and this study have assumed an allowance for water of 15 pounds per crew member per day. Six pounds of this water would be used by the crewman to reconstitute his food and to drink. The balance of the daily allowance (9 pounds) would be used for cleansing and body care. Figure 11 shows that this daily water requirement represents a total weight for each mission of from 24,000 to 84,000 pounds. It is possible to recover all of this water; however that which is contained in the fecal matter is small and it is neglected. There is an offsetting compensation to this neglect. It is planned to supply the Skylab I and these studied mission crews with a variety of foods from freeze-dried, through conventionally frozen (TV dinners), and including fresh fruit and leaf vegetables. The fresh and frozen foods carry a large percentage of water, and this water would more than compensate for that lost as fecal

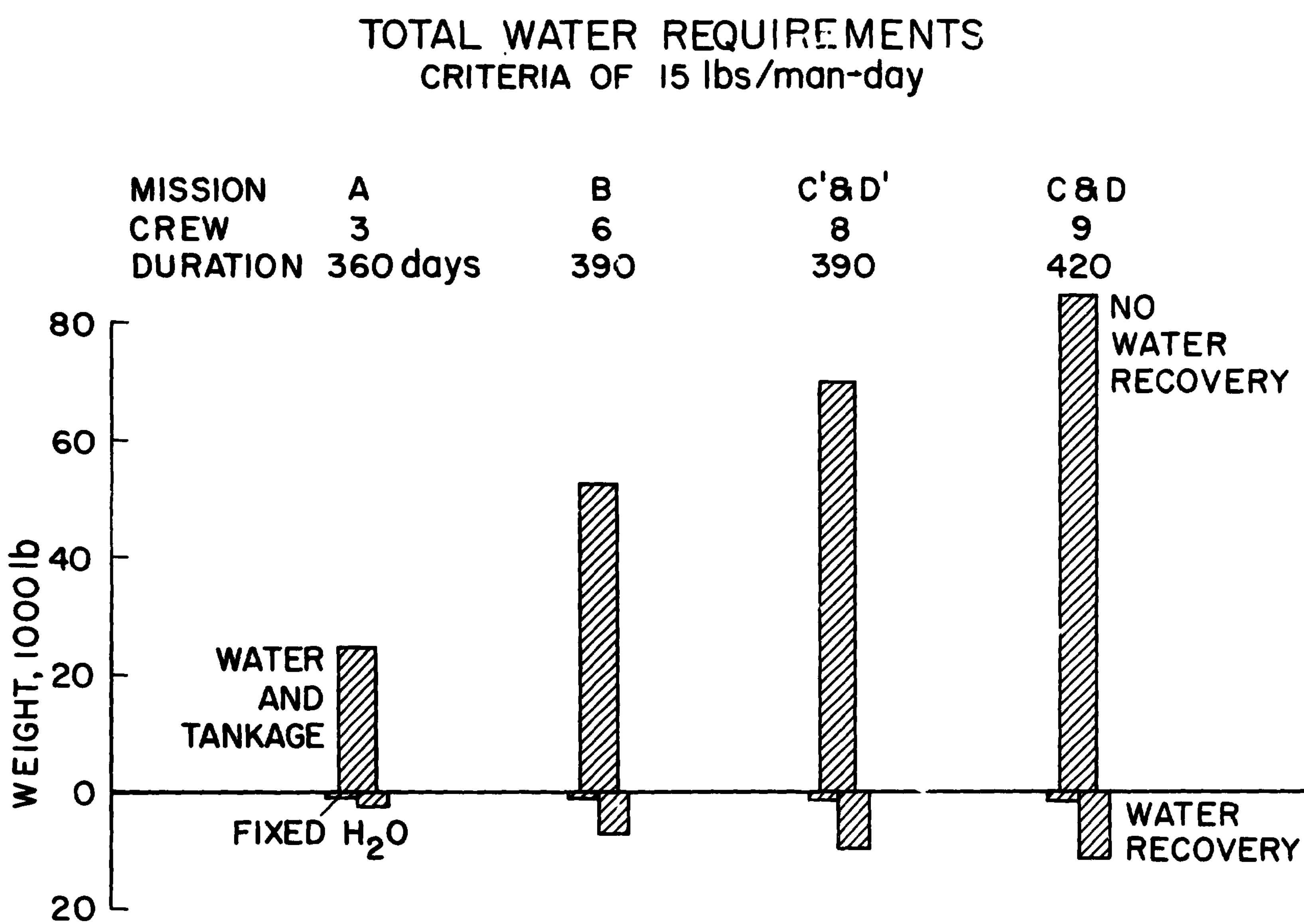

TOTAL WATER REQUIREMENTS
CRITERIA OF 15 lbs/man-day
MISSION A B C'&D' C&D
CREW 3 6 8 9
DURATION 360 days 390 390 420
WEIGHT, 1000 lb
80
60
40
20
0
20
WATER AND TANKAGE
FIXED H_2O
NO WATER RECOVERY
WATER RECOVERY

Figure 11.

water. A recovery of any portion of the daily water used could represent real savings in logistic supply loads.

Several water recovery methods have had extensive tests under almost in-flight conditions. These methods include air evaporation, vapor diffusion, and reverse osmosis, and probably of these, the most highly developed is the air evaporation type. This type has the least difficulty in recovery of either wash, urine, or air condensate water but it does require frequent wick replacement. Each of the types however, need actual in-flight tests to better determine which type of used water each recovers the best. Because of these development needs there would be water recovery systems as experiments during each mission.

An air evaporative water recovery system would be an experiment on Mission A, and it is expected that it should be sufficiently developed to be a spacecraft system for Mission B. For subsequent missions the vapor diffusion and reverse osmosis would be serially flight tested, qualified and installed as spacecraft systems. Each of these water recovery techniques have certain desirable characteristics and each should be tested and developed so as to be available for the Space Station. The net savings to Mission B, C and D from the recovery of water would be great. If the condensate, urine and wash water is recovered, the savings would vary from about 40,000 pounds for the early mission to 70,000 for Mission D, Figure 11. The effects of these savings upon the water supply system in terms of the spacecraft and the logistic requirements will be summarized at the end of this section.

Data Collection and Communications

The data collection and communications systems for the interim space stations will be primarily extensions of the Skylab and Apollo systems; however reconfigured to handle the larger crew sizes, more operational activities and greater experiment investigations. In this section, the methods available to handle the data and communications are discussed, the communications requirements are stated, and the critical link performances are evaluated. Since it is assumed that these spacecraft will utilize the Manned Spaceflight Network (MSFN) for all communications, the coverage and capability of the MSFN stations are discussed, and postulated data links are evaluated for adequacy to carry the anticipated data load.

The average data load generated by the experiments and by the interim space station systems will be on the order of 10^{11} bits per day. This data load is approximately an order of magnitude greater than that expected from the Skylab program and two orders of magnitude greater than that from Apollo, Refs. 12 and 13. This increase does not require a commensurate increase in communications equipment complexity and weight. This additional data transmission will be accomplished as on the Skylab by means of wideband links to convey primarily the experimental data. The major effect of the increased data load will be felt in the increased data processing equipment carried by a lab. The secondary effect will be the need for increased data recording and handling capability at the ground stations.

It is expected that the experiment data will have some onboard preliminary processing, recording or interpretation. The nature of this processing will vary according to the mix of experiments being performed, and this procedure might be changed at any time during the experiment's life. The use of scientifically trained personnel onboard the station permits "on the job" experiment and processing modifications and should maximize the scientific, sociological or economic benefits from these data. It also permits a screening of the data to sift out that which is relevant and to minimize the irrelevant data load sent to Earth. It is not anticipated that this would be a closed process but that, by means of voice and TV channels, scientific team members on the ground and in space could be interconnected for near real time experiment control. However, the data which is received by MSFN would be sent to a central processing facility, perhaps initially at Goddard, for the usual processing before it is distributed to the principal scientific investigators.

Link Requirements - Communications links are required between the Apollo CSM crew, the DWS crew and the MSFN as shown in Table 15. These links are similar to those required for the Skylab Program, but with the additions indicated. It is proposed to have a teletypewriter in the airlock module, which sends data by means of the command link. A backup unit would be installed in the DWS as well.

Table 15
Communications Link Requirements

DWS-CSM	DWS-MSFN	CSM-MSFN
Two-way voice	Two-way voice	Two-way voice
Ranging	Ranging/tracking	Ranging/tracking
	Housekeeping T/M	Housekeeping T/M
	* Two-way TV	
	* High Data Rate Scientific T/M	Command (Uplink)
	Teletype (Uplink)	
	Command (Uplink)	

* Additions to Skylab Requirements

The interim station must have the capability both to send and to receive television so as to permit detailed experiment conferences with the ground. It is proposed to locate both the television camera and the transmitter aboard the DWS. Although the CSM transmitter is available for this purpose, and would serve as backup, its regular use entails the activation of circuits on the quiescent stored, docked CSM. It is operationally simpler to have the entire TV system within the space station, and this permits greater flexibility in the TV system. Provisions will be made for connection of the television camera in various sections of the MDA, AM, and DWS as well as the CM and then route signals to the DWS transmitter. It is intended to have the TV antenna, receiver and display located in the DWS. This makes the Apollo CSM TV system an entirely independent but available communications link. The availability of two-way television gives the interim stations much more communication system flexibility and capability than is currently planned on the Skylab. The addition of high data rate telemetry to the interim station's system will be discussed in the following section.

Spectrum Restrictions and Channel Capacity - The greater scientific data load and the need for frequent readout of the scientific data accumulated during these interim missions cannot be handled by existing Apollo-Skylab configurations. It is necessary to have some additional wideband data handling capability links added to the DWS systems.

The MSFN radio receivers are continuously tunable over the range of 2200 to 2300 MHz. The interim station telemetry is considered to be confined to this band and the subsequent data transmission performance will be assessed on this basis. Other bands, such as 1435 to 1535 and 1700 to 1710 MHz could be available if the STADAN facilities were to be used but these have been assumed to be fully committed to other programs. In addition, laser and/or millimeter wave links could be employed as experiments with the DWS for their assessment as high data rate links. This interim program would be expected to implement data transmission experiments employing possible future type links in order to evaluate their applicability to the space station. However, they have not been considered in this study in order to evaluate the Apollo-Skylab system capability with currently available or minimally modified equipment.

The current frequency allocations in the 2200 to 2300 MHz band are delineated in Table 16, Refs. 14, 15, and 16. It has been assumed that some lunar programs, manned or unmanned, will be underway during the flight periods of Skylab and through the interim space stations; so the 2275 to 2285 MHz band has been reserved for these lunar purposes. The Earth resources program was assumed to be continuing and that it would employ the currently stated ERTS frequency assignments. Also indicated in Table 16 is a suggested plan for the interim stations' system band usage. A 10 MHz band centered at 2295 MHz would be available for transmission of television signals and low data rate telemetry, biomedical data, two-way voice and emergency channels. This band appears to be more than adequate when the requirement for two-way television has been considered to mean TV capability present both in the orbiting lab and on the ground, but not simultaneous transmission.

The ranging transponder is necessary for the logistic docking maneuvers and requires about 4MHz bandwidth. It can operate at any frequency in the 2200 to 2300 MHz band. Ranging is only needed for a few hours through launch, orbital insertion and docking. During this period, the ranging signal could preempt one high data rate telemetry channel, for during these crew change procedures, the experiments would be at a low level of effort, and have a reduced need for transmitting data.

Table 16
MSFN Downlink Spectrum Utilization

Frequency MHz	
* 2300 2290	10 MHz Interim Space Station TV and Housekeeping T/M
2290 2285	ERTS Housekeeping T/M (2287.5 Apollo Command Module PM Carrier)
2285 2279.5	Uncommitted (2282.5 Apollo SIV-B PM Carrier)
2279.5 2275.5	ALSEP (2277.5 Apollo SIV-B FM Carrier)
2275.5 2255.5	ERTS Return Beam Vidicon Data Downlink (2272.5 Apollo Command Module FM Carrier)
* 2255.5 2239.5	16 MHz Interim Space Station Data Downlink
2239.5 2219.5	ERTS Scanner Data Downlink
* 2219.5 2200	19.5 MHz Interim Space Station Data Downlink

* Indicates Interim Space Station Channels

The foregoing stated commitments leave available for station use two channels, one centered at 2210 and one at 2247.5 MHz for the transmission of real time or stored scientific data. To use these bands, it requires the addition of two transmitters and antennas in the DWS. By use of the 2210 and 2247.5 MHz channels, it is possible to transmit two data streams of 19.5 and 16 Mbps respectively, in either PCM/PM or PCM/FM. Current design practice uses a lower ratio of bit rate to bandwidth; however a ratio of 1:1 should be technically feasible and in current practice by 1975. This ratio, with a sufficient S/N ratio, is appropriate to a bandwidth limited rather than power limited link. Scheduling of transmissions should resolve, if there are any, conflicts between the Lunar ALSEP and the SIV-B stage housekeeping telemetry. The frequency congestion which could result from concurrent flights of ERTS and Apollo may be relieved by a shift in the CSM transmitter frequency to the 2290 to 2300 band as previously mentioned.

Data Rate Requirements versus Capability - The adequacy of the proposed downlink telemetry channels was examined in terms of transmitter power as well as in terms of mission experiment requirements. The television and other telemetry requirements are basically the same as that for Apollo and which already have demonstrated capability considerably beyond the needs outlined for the interim stations; there is thus no need to verify their performance margins. Similarly, the uplinks designs, operating in the 2090 to 2120 MHz band, can be shown to be more than satisfactory.

A detailed examination of the downlink telemetry power budget was made. The details are presented in Appendix C. These results show that a transmitter on the order of 12 watts is satisfactory for a wideband channel and that it would be capable of transmitting approximately 20 million bps. If a performance channel of less than 16 million bps were required, it would need only slightly less power. This transmitter power level is consistent with some of the Apollo and Saturn IV-B equipment used with the Apollo lunar program. The two additional transmitters required for the two wideband data channels increases the interim station power requirements by less than 200 watts.

The communications system requirements were also examined in terms of the ground reception equipment and considering only the MSFN 30 foot diameter antennas as being available. The details of this evaluation are given in Appendix D. These results showed that the station coverage of continental U.S. is satisfactory for orbits of 28 1/2° inclinations. However, for the missions with an orbit of 50°, some supplemental coverage may be necessary.

In order to estimate the total data which may be regularly and periodically returned to Earth, one must evaluate the number of times the space station overflies ground stations on a regular basis. If a direct daily overflight of a single site is assumed, we can compute an upper bound for this daily data return capability to Earth. A very detailed study would be necessary if it were necessary to evaluate the effects such as experiment schedules, orbit inclination and drift and site masking pattern so as to determine the actual transient data transmission time available and subsequently the recorder capacity required. On an average basis, the data transmittal capability to Earth should exceed or at least equal the data that is possible to be generated by the experiments. Figure 12 shows the station data transmission rate capability for one direct overfly of one ground station each day. Also shown in this figure is the average daily data load generated on the spacecraft for each of the missions. It is evident that the proposed links would be more than satisfactory for the conditions of one daily overflight. One direct overflight of a ground station might not be made each day: however, the results in Appendix D show that the net daily overflights within range of the ground stations are at least equal to or greater than one direct overflight. The network linkup of ground stations in the MSFN should have data transmission capability at least equal to that for the single reception site.

Should the experiment program schedule be revised, the data loads could increase markedly. The proposed channels then might not be adequate. This overload could be reduced in several ways: by increased astronaut screening of data prior to transmission, increased logistic hard copy or tape data return, addition of experimental channels useage, or preemption of other channels such as the ERTS S-band channel. Since the ERTS satellite occupies a near polar orbit while the manned stations are a lower inclination orbit, the interference between these two spacecraft

COMPARISON OF INTERIM SPACE STATION DATA GENERATION AND RADIO LINK TRANSMISSION RATES

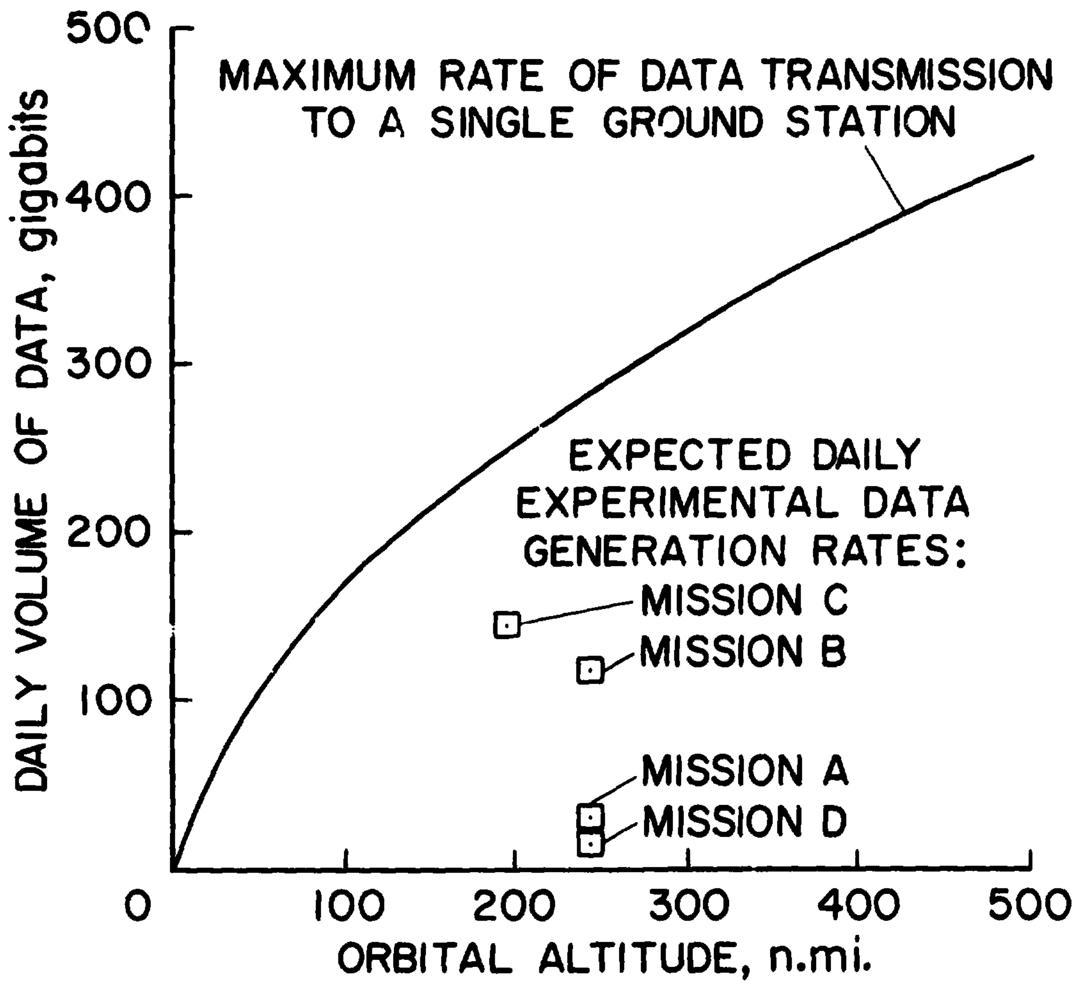


Figure 12.

simultaneously using the same ground station should be at a minimum. On the other hand if an ERTS is not in orbit its channel could be used, and it does give a margin for growth to the interim space station communications system.

Radiation and Micrometeoroid Protection

Radiation and micrometeoroid protection are considered together for they are both passive systems and they are each designed to sustain the crew's safety against a probable event. The probability that either a solar flare will occur or that the spacecraft will be bombarded by micrometeoroids are high. These passive systems are thus vital systems for the interim space stations.

Radiation Environment - How much energetic particulate radiation each human can stand is not explicitly defined for it is dependent upon the energy of the particles as well as the tolerance of each individual. For this study, a dose limit to the blood forming organs of 40 rem (Roentgen equivalent man) per six month period has been assumed as the criteria. This is the same criteria as was used for the recent NASA Space Station/Base studies, Ref. 17. As is often done in shielding requirement determinations, it has been assumed that the radiation absorbed dose, designated rad, is equivalent to the biological radiation damage or Roentgen equivalent man. The shielding determinations are made upon the protection required for the human blood forming organs. The body shields these organs, and this shielding effect is conservatively estimated to be equivalent to 5 grams/$\overline{cm}^2$ of aluminum. If only the primary structure and the micrometeoroid shielding is considered, the effective shielding furnished by these are estimated to be equivalent to 3 grams/$\overline{cm}^2$ of aluminum. To be conservative in the shielding determinations, any shielding effect which could be gained by selectively locating equipment around the spacecraft's walls has not been included. These foregoing assumptions and criteria were the bases used in determining the amount of radiation shielding required.

The various contributors to the radiation environment for the interim space stations are shown in figures 13 through 15, Refs. 17, 18, and 19. Figure 13 gives the trapped electron dose per year behind a 1 g/cm^2 aluminum shield as a function of orbit altitude for orbit inclinations of 30°, 60° and 90°. Figure 14 presents similar types of data for trapped protons. Figure 15 presents the solar proton dose per year, during a year of average maximum solar activity, in terms of shield thickness in g/cm^2 of aluminum and for a 50° inclined orbit. These data are predicated on a probability of 0.99 of not accumulating more than the indirect dosages. At orbit inclinations less than about 35°, the Earth's magnetic field effectively shields the spacecraft from solar protons. Figures 16 and 17 show the total integrated dose in rad/year as a function of radiation shield unit weight in g/cm^2 of aluminum for discrete altitudes from 150 to 500 n.m. These figures are derived from data which show the effectiveness of shield thickness against both trapped electrons and trapped protons, Ref. 18. Figure 16 is for inclinations between 45° and 55°, and figure 17 is for inclinations between 25° and 35°. At inclinations less than about 35°, the radiation shield need only protect against the trapped radiation; while above this inclination it must protect against a portion of the solar flare radiation as well. If it is assumed that each crew member can tolerate only 80 rad/year total dose, the radiation shielding which must be added to the 4000 square feet of the spacecraft external surfaces can be readily determined from these figures for each mission.

Meteoroid Shielding - For the flight durations of these missions, some micrometeoroid shielding other than that furnished by the basic spacecraft structure will be necessary. Figure 18 shows the specific micrometeoroid shield requirement as a function of mission duration, based on a probability of 0.99 that no penetrations of the manned spacecraft will occur, Refs. 20 and 21. The meteoroid shield type is assumed to be two aluminum sheets separated by a polyurethane foam energy absorber. The inner sheet is the primary spacecraft structure. Thus the meteoroid shield weight consists of that contributed by the outer aluminum sheet and by the polyurethane foam.

The various portions of the interim space station which would require

TRAPPED ELECTRON DOSE AS A FUNCTION OF ORBIT ALTITUDE

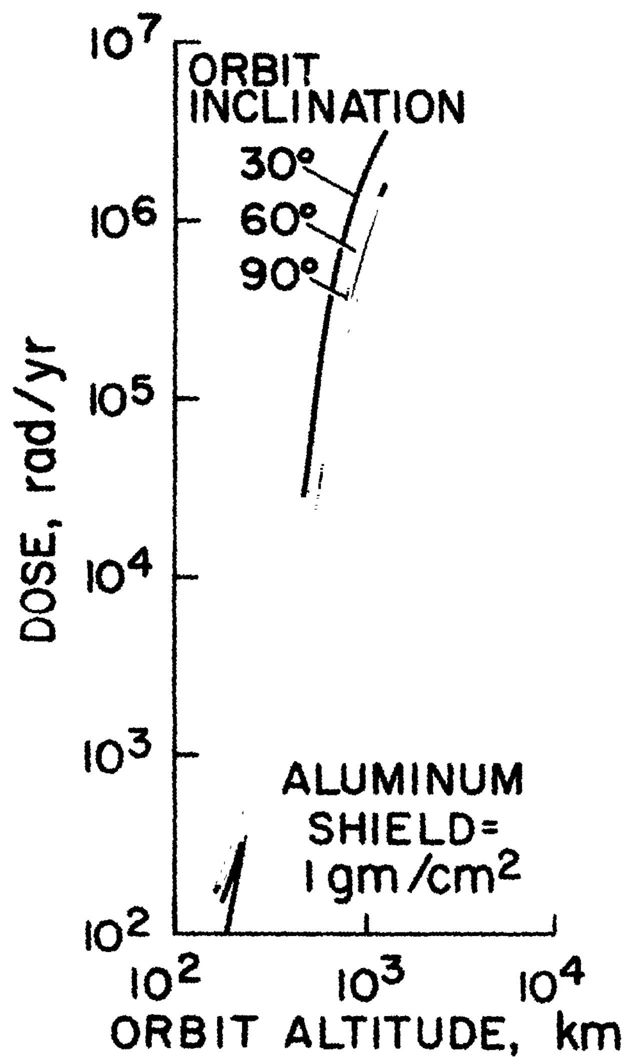


Figure 13.

TRAPPED PROTON DOSE AS A FUNCTION OF ORBIT ALTITUDE

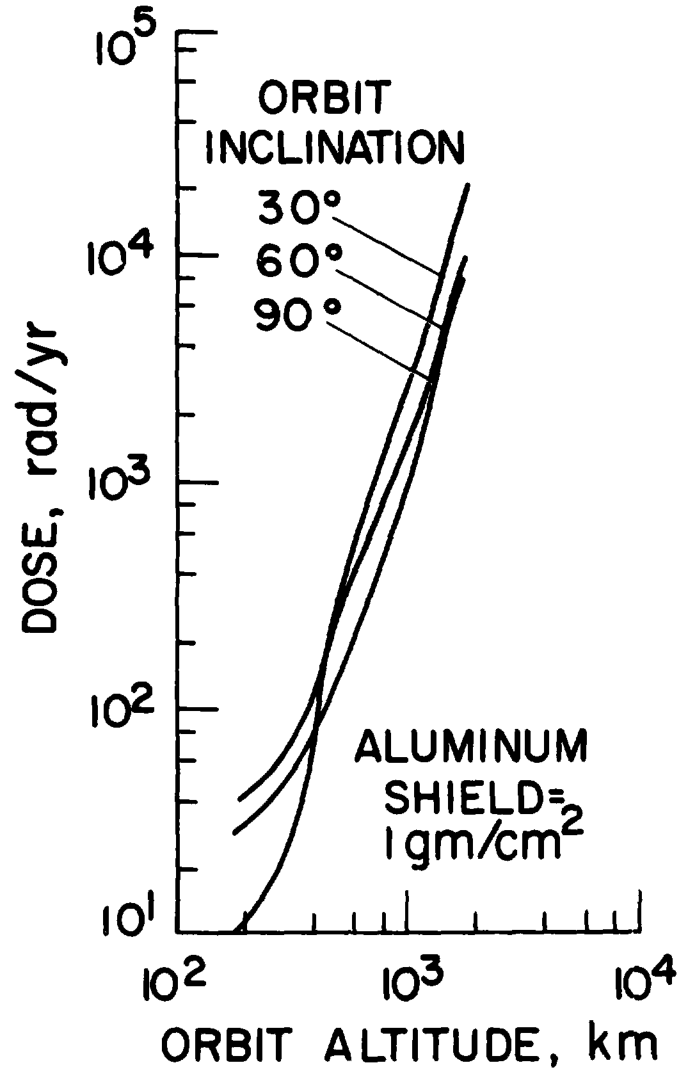


Figure 14.

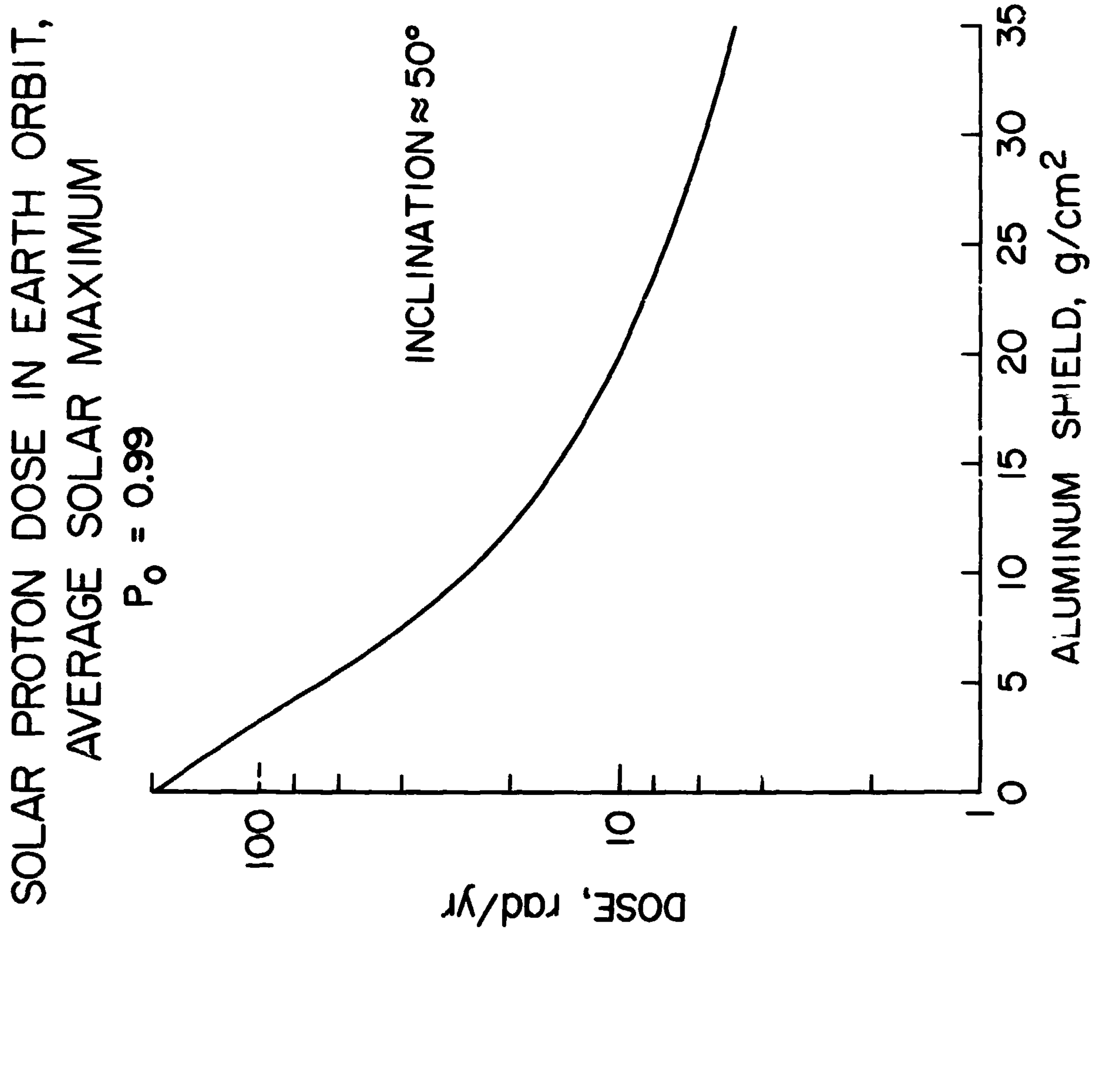

SOLAR PROTON DOSE IN EARTH ORBIT,
AVERAGE SOLAR MAXIMUM
Po = 0.99
INCLINATION ≈ 50°
100
10
1
DOSE, rad/yr
0
5
10
15
20
25
30
35
ALUMINUM SHIELD, g/cm2

Figure 15.

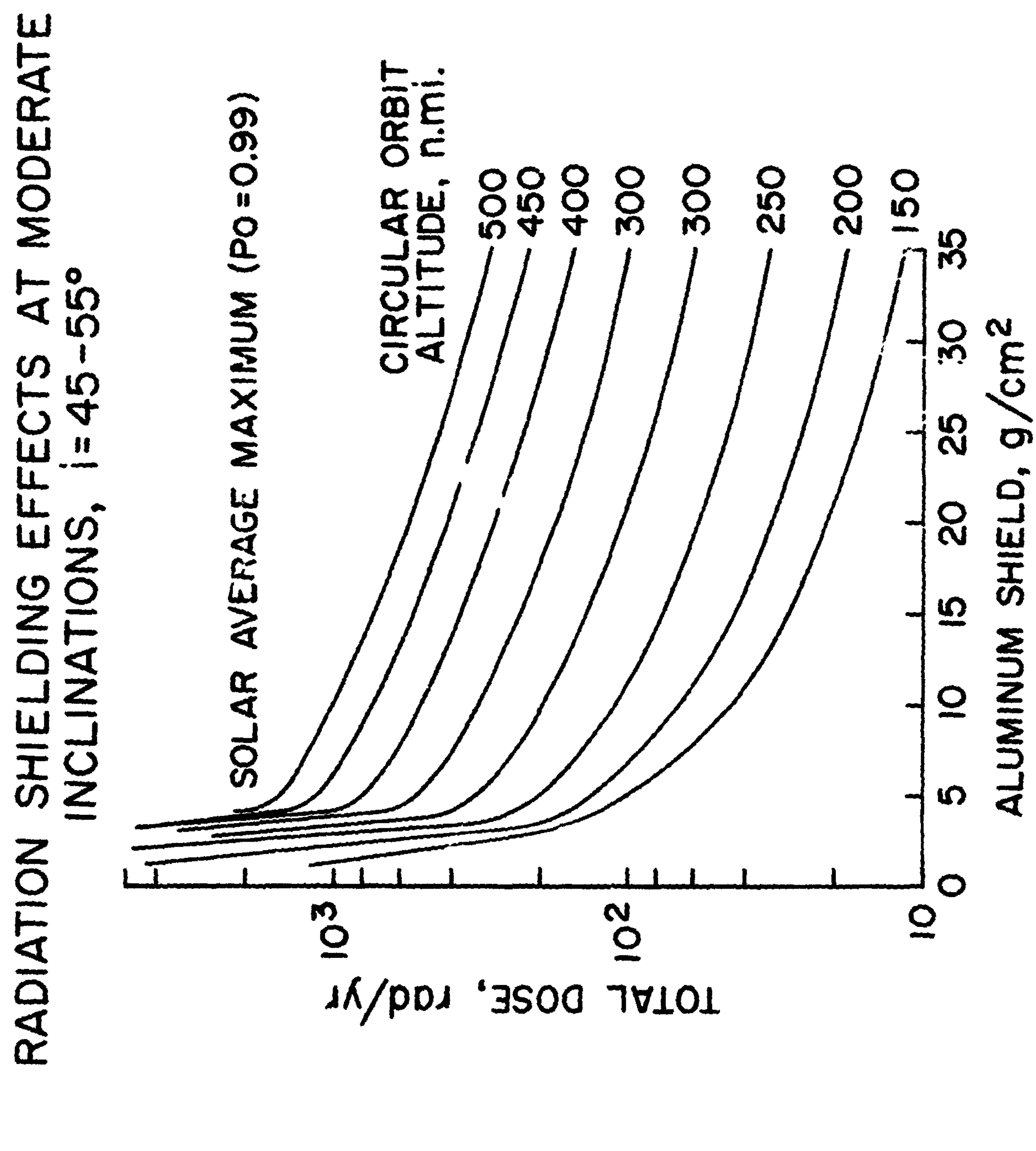

RADIATION SHIELDING EFFECTS AT MODERATE INCLINATIONS, i=45-55°
SOLAR AVERAGE MAXIMUM (Po=0.99)
CIRCULAR ORBIT ALTITUDE, n.mi.
500
450
400
300
300
250
200
150
TOTAL DOSE, rad/yr
10
10^2
10^3
0
5
10
15
20
25
30
35
ALUMINUM SHIELD, g/cm^2

Figure 16.

RADIATION SHIELDING EFFECTIVENESS AT LOW ORBIT INCLINATIONS, i = 25-35°

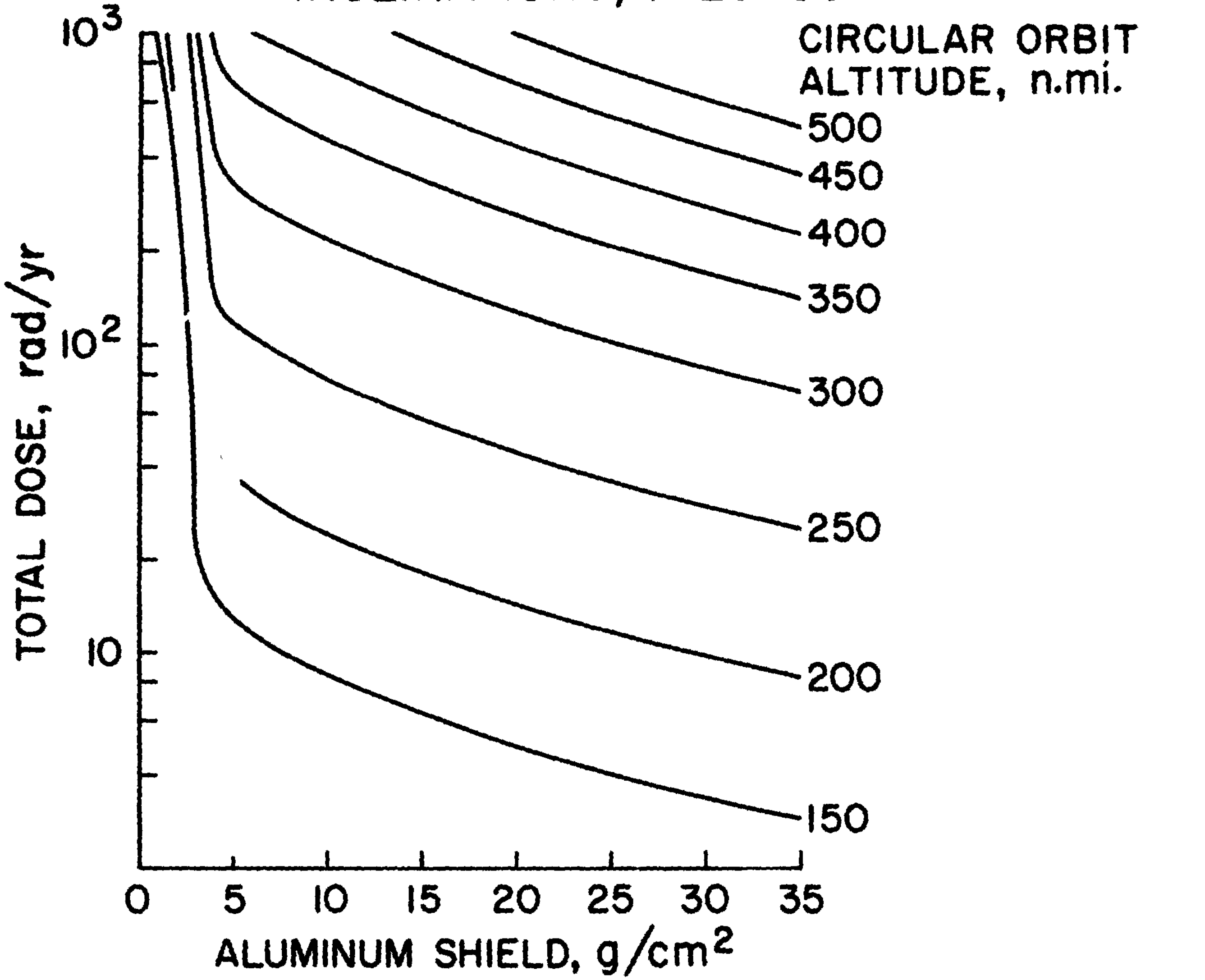


Figure 17.

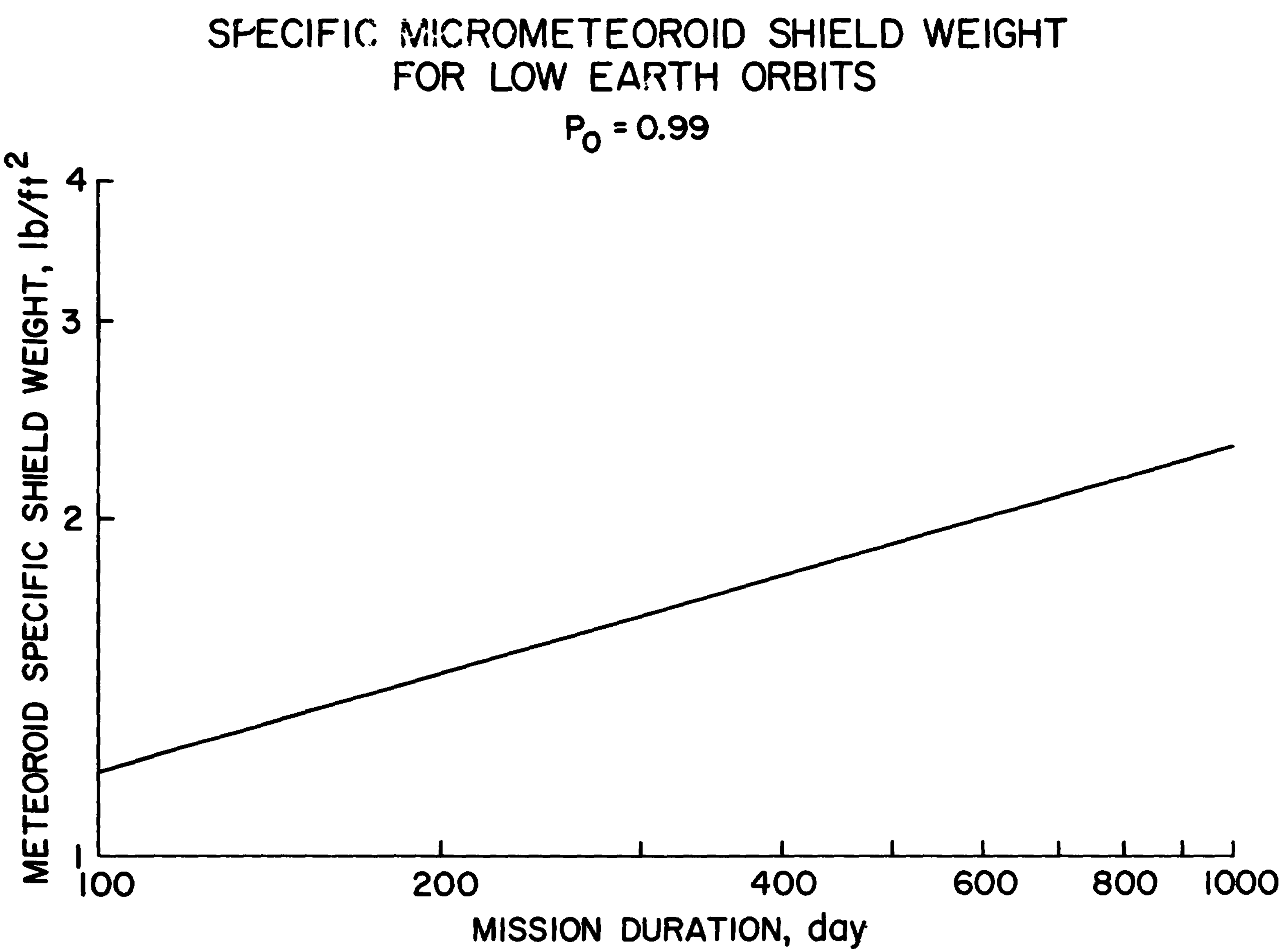

SPECIFIC MICROMETEOROID SHIELD WEIGHT
FOR LOW EARTH ORBITS
$P_0 = 0.99$
METEOROID SPECIFIC SHIELD WEIGHT, lb/ft²
4
3
2
1
100
200
400
600
800
1000
MISSION DURATION, day

Figure 18.

protection and the surface area to be protected are as follows:

Multiple Docking Adapter (MDA)	803 ft^2
Airlock Module from MDA to fixed SLA shroud	1207
Cylindrical portions of Workshop	1520
Total Area	3520 ft^2

The storage tanks mounted at the lower end of the workshop are considered to be heavy enough to furnish their own meteoroid protection as well as to furnish protection for this end of the spacecraft. The fixed SLA shroud and enclosed tanks give some of the needed protection for the airlock module. Figure 19 shows the total meteoroid shield weight required for the interim station as a function of mission duration. For a one year mission, this shield weight is over 6000 pounds. When this is added to the weight of the spacecraft primary structure, it gives an average total wall density of about 3 grams/cm^2.

If the mission durations should be extended beyond their designed duration, figure 20 shows the effect for the added duration upon the probability of meteoroid puncture. The three curves shown are for the four Missions A through D with design durations of 360, 390, and 420 days respectively. Extending the use of the Mission A spacecraft for six months would increase its probability of meteoroid puncture by one-half percent. The possibility of space station meteoroid puncture would need be given consideration in any planning for post program use of spacecraft by reactivation.

In determining the amount of materials required for both the micro-meteoroid and the radiation shields, careful consideration was made for the contribution that the former could make on the latter's needs.

Orbit Maintenance

The atmosphere is sufficient, even at altitudes greater than 200 nautical miles, to cause the space station's orbit to decay appreciably. The atmospheric drag is greatest upon the sail like solar cell arrays. In order to maintain a desired orbit altitude, a drag makeup system is required and it is included as one of the station systems. This system consists of thrusters, plumbing, controls and the fuel and its tankage.

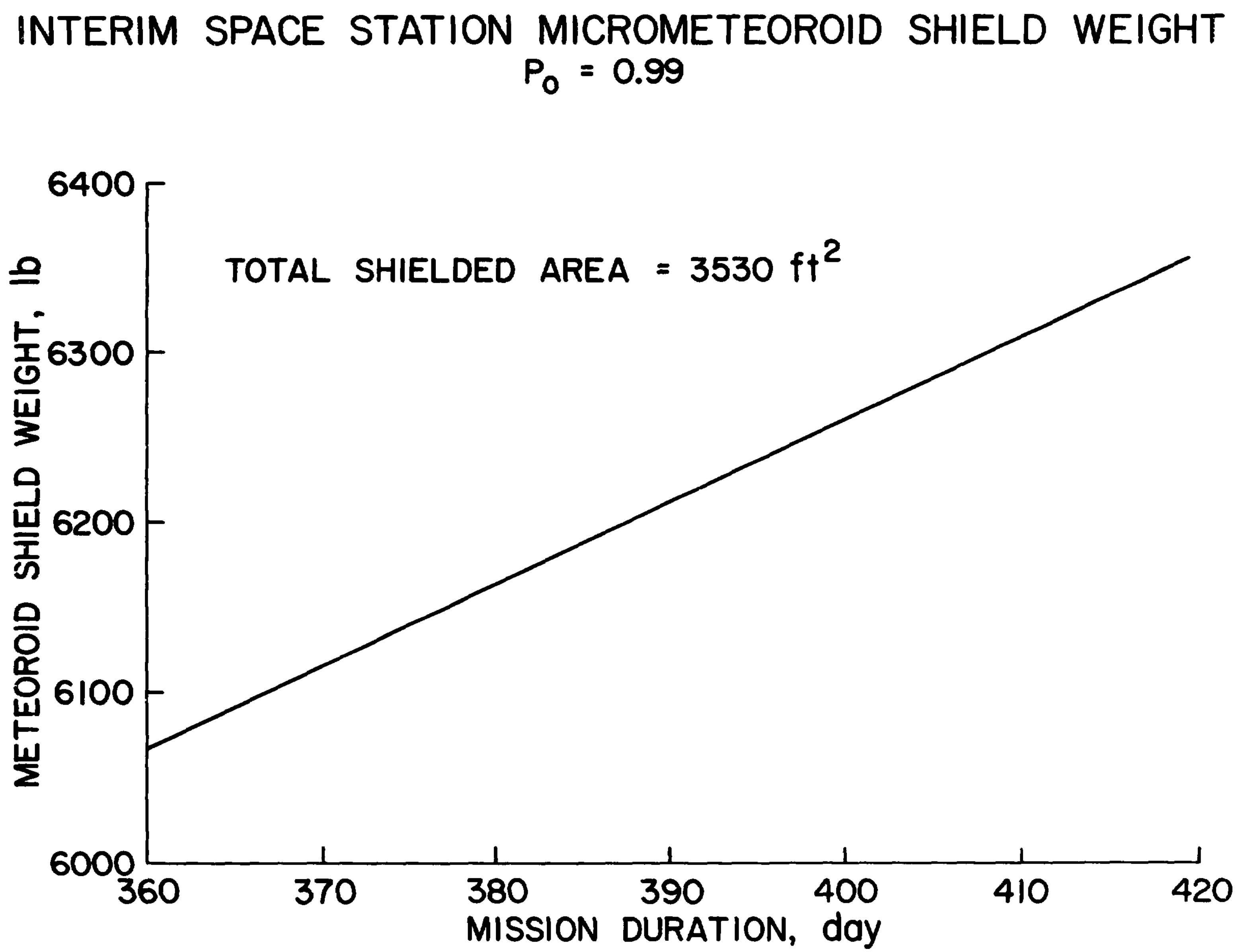

INTERIM SPACE STATION MICROMETEOROID SHIELD WEIGHT
P_o = 0.99
TOTAL SHIELDED AREA = 3530 ft^2
METEOROID SHIELD WEIGHT, lb
6400
6300
6200
6100
6000
360
370
380
390
400
410
420
MISSION DURATION, day

Figure 19.

SPACECRAFT METEOROID PUNCTURE PROBABILITIES
FOR EXTENDED MISSION LIFETIMES

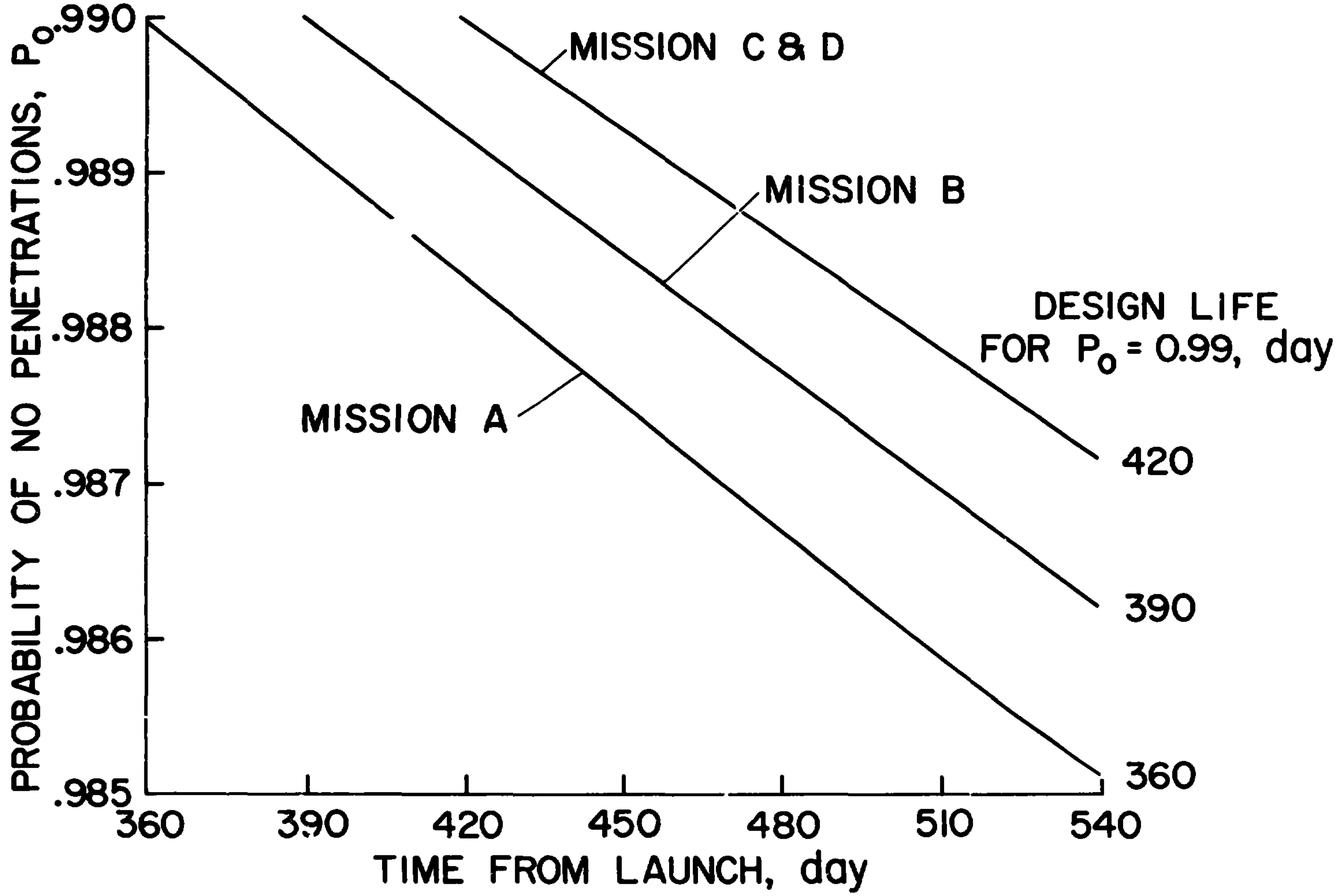


Figure 20.

The analysis to determine the drag of tl here upon the space station was performed in two parts: that for the space station alone, and that for the solar arrays alone. This was done in order to see the effect of the types of and power system sizes chosen on the drag makeup system required. Experience has shown this to be acceptable, for the arrays are usually separated as on booms from the spacecraft proper. The solar panel analysis depended heavily upon the results reported in Ref. 22. This report shows that when both the electrical power system (including arrays, batteries and controls) and the drag makeup system (fuel, tankage and thrustors) are considered, and for altitudes greater than 185 nautical miles, then their combined specific weight is lower for rrays which are sun-oriented than for arrays which are orbit-oriented. Since none of the missions considered in this study were anticipated to require altitudes less than 190 nautical miles, the analysis which follows considered only sun-oriented arrays. Sun-oriented arrays mean that the array. are kept perpendicular to the sun at all times. Figure 21 which is from Ref. 22, and it is for the solar arrays only, shows the specific weight for the drag makeup system (fuel, tanks, and thrustors) as a function of orbit altitude. The values shown in this figure are for a one-year mission and for a drag makeup fuel having an I_{sp} of 320 seconds.

The effect of atmospheric drag upon the space station is determined in a different manner. In order to calculate the drag makeup fuel weight, it is necessary to know the station's effective m/C_DA as well as the fuel and orbit characteristics. The following station parameters were used in this determination:

m = 7000 slugs
A = 380 Ft^2
C_D = 2
I_{sp} = 320 seconds
Mission = 1 year duration

And the space station includes the workshop, MDA, AM, and the docked Apollo CSM. From these assumptions, it can be shown analytically (the detailed analysis is given in Appendix E at the end of this report) that the velocity impulse required per orbit for drag makeup is:

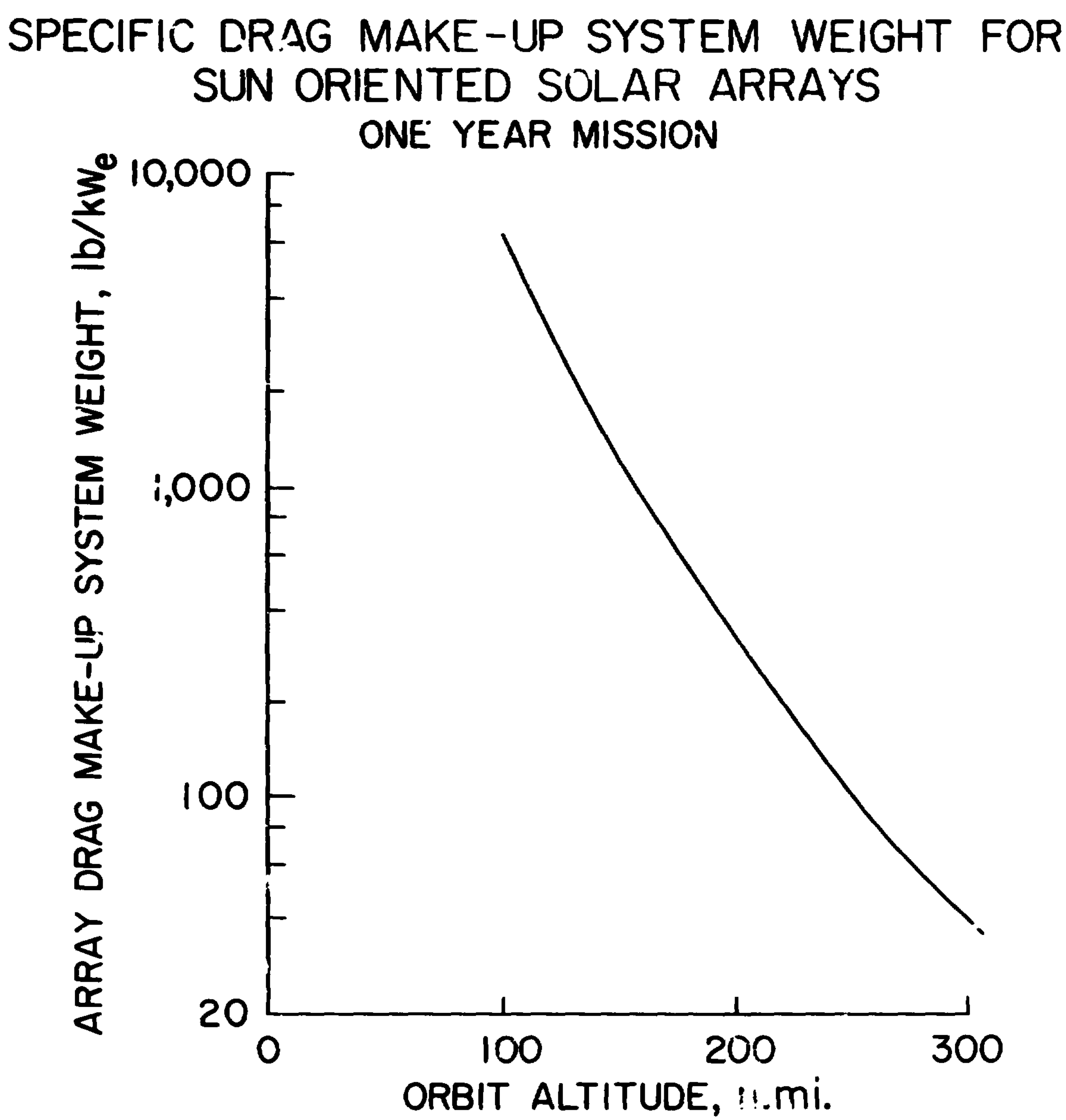

SPECIFIC DRAG MAKE-UP SYSTEM WEIGHT FOR
SUN ORIENTED SOLAR ARRAYS
ONE YEAR MISSION
ARRAY DRAG MAKE-UP SYSTEM WEIGHT, lb/kWe
10,000
1,000
100
20
0
100
200
300
ORBIT ALTITUDE, n.mi.

Figure 21.

$$\Delta V = \frac{V_c}{3} \cdot \frac{\Delta P}{P}$$

where

V_c is circular velocity at the initial altitude

ΔP is change in orbit period due to the drag

P is period of the initial orbit.

The dimensionless quantity $\Delta P/P$ has been determined for a range of altitudes and for given values of m/C_DA in Refs. 23 and 24. Using these data, the drag makeup fuel required for each orbit of the space station is then:

$$m_f = m_{total} \left(1 - e^{-\Delta V/g_o I_{sp}}\right)$$

If the space station drag fuel requirement for one year is integrated with the drag makeup system and fuel for the solar arrays from Figure 21, then figure 22 results. This shows the drag makeup system weight for the entire space station at six discrete power levels from 1 to 25 kw for a range of orbital altitudes. Also shown as the lowest curve is the drag makeup system weight for the space station oriented end-on to the flow and without solar arrays. (Power = 0)

As was discussed in the previous section in Figures 16 and 17, the radiation shielding requirements increase with increases in orbit inclinations and altitudes. There is an interesting tradeoff which may be made when the shielding weight requirements are combined with the drag makeup system whose weight decreases with altitude. The altitude, for a given inclination, at which the sum of these two systems is a minimum should be of interest. Figures 23, 24 and 25 show these combined systems weights as a function of orbit altitude and for the power levels corresponding to those for missions A through D. The drag makeup system weight, depending on the orientation of the solar cell panels, decreases nearly exponentially with altitude up to the altitude at which the spacecraft structure, meteoroid shield, and the human body can furnish the radiation shielding necessary for the human body blood-forming organs. This is the knee of the curves shown. Above this altitude, additional radiation shielding is required. The figure shows a minimum weight at an altitude of interest for these missions. For a space station inclination of 28.5 degrees, the orbit altitude of 245 n.m. has the minimum weight; and at 50 degrees inclination, the optimum altitude is 195 n.m. If these results

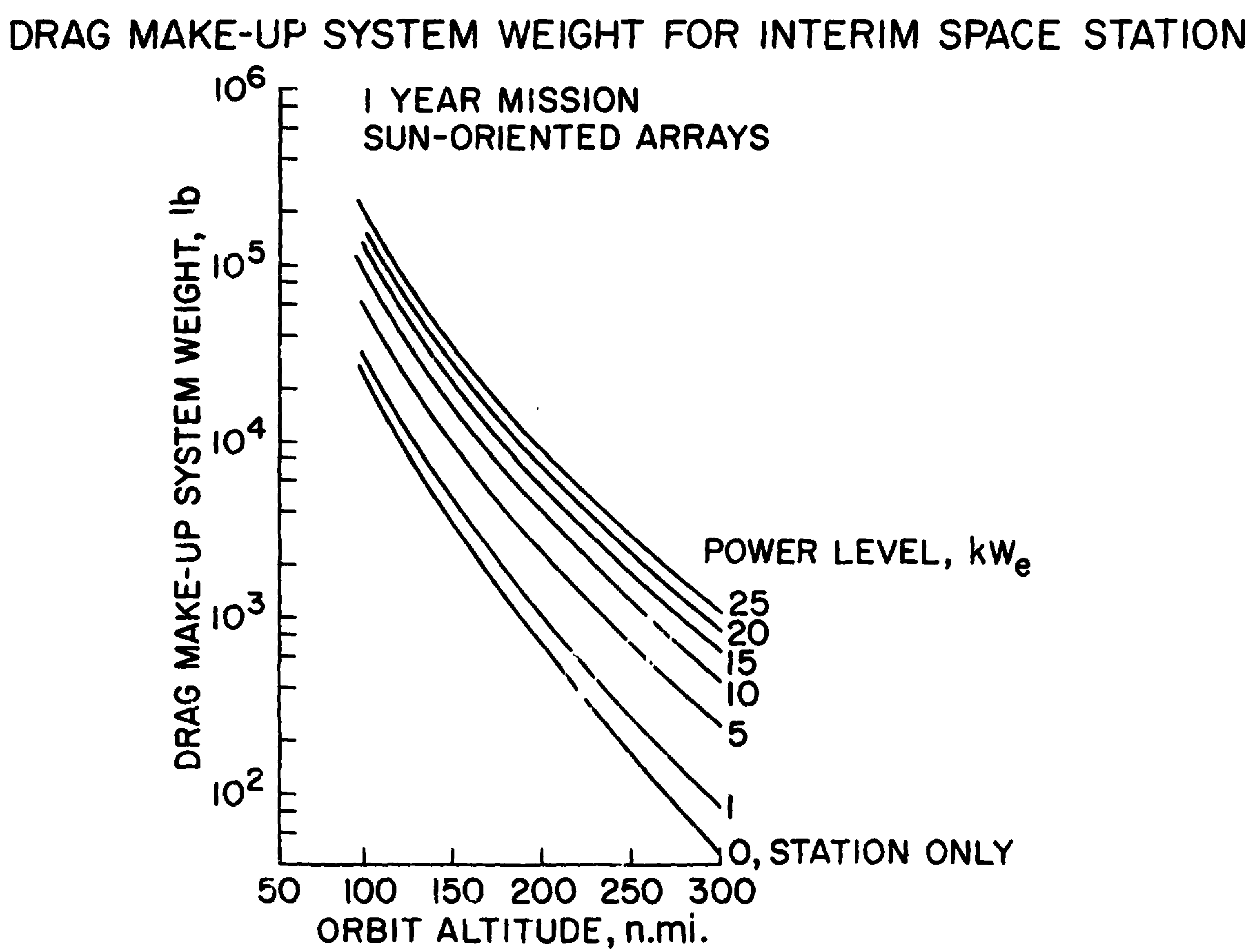

DRAG MAKE-UP SYSTEM WEIGHT FOR INTERIM SPACE STATION
I YEAR MISSION
SUN-ORIENTED ARRAYS
DRAG MAKE-UP SYSTEM WEIGHT, lb
10^6
10^5
10^4
10^3
10^2
POWER LEVEL, kW_e
25
20
15
10
5
1
0, STATION ONLY
50
100
150
200
250
300
ORBIT ALTITUDE, n.mi.

Figure 22.

DRAG MAKE-UP AND RADIATION SHIELD SYSTEMS WEIGHT
FOR MISSION A

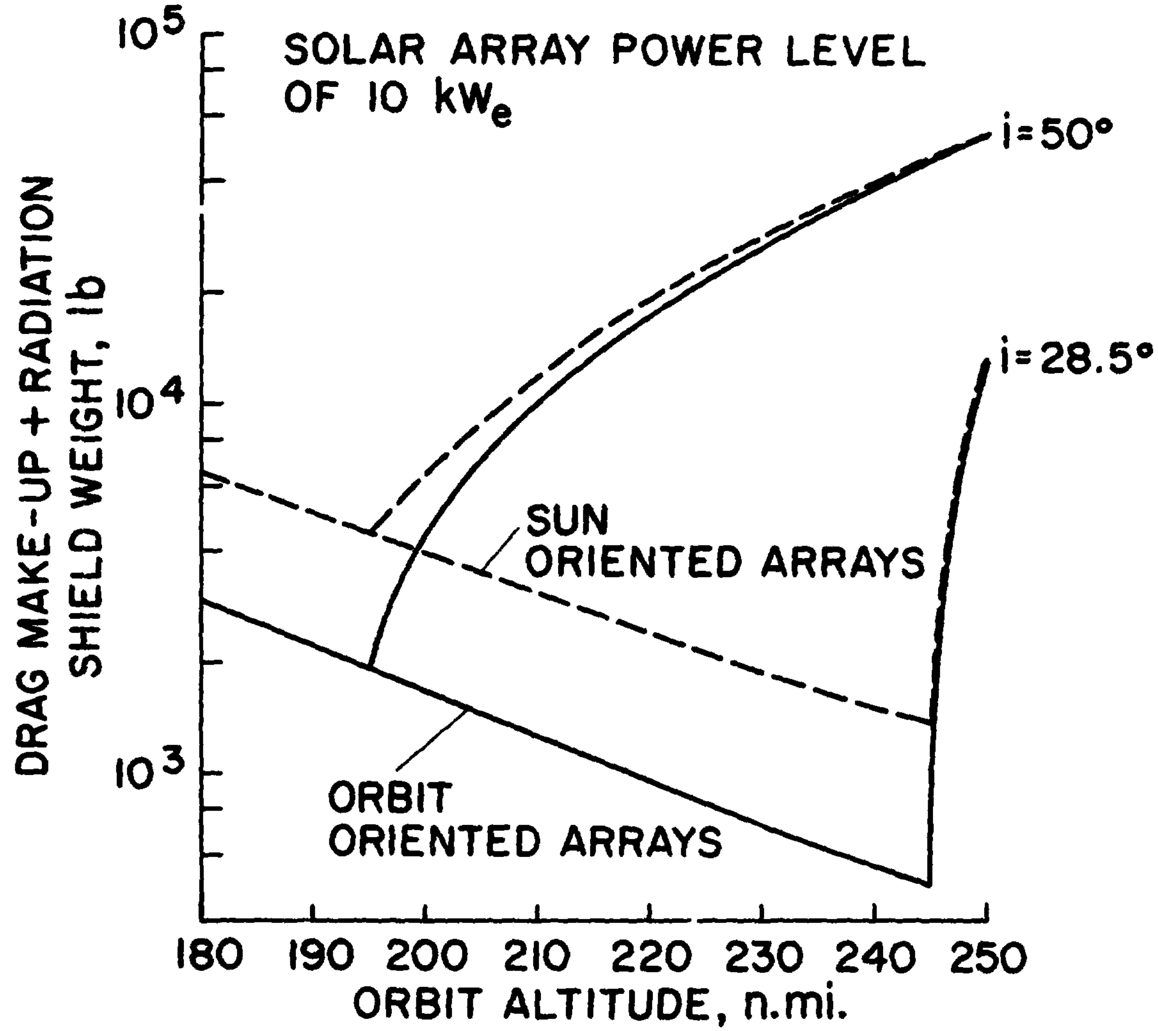


Figure 23.

DRAG MAKE-UP AND RADIATION SHIELD SYSTEMS WEIGHT
FOR MISSION B

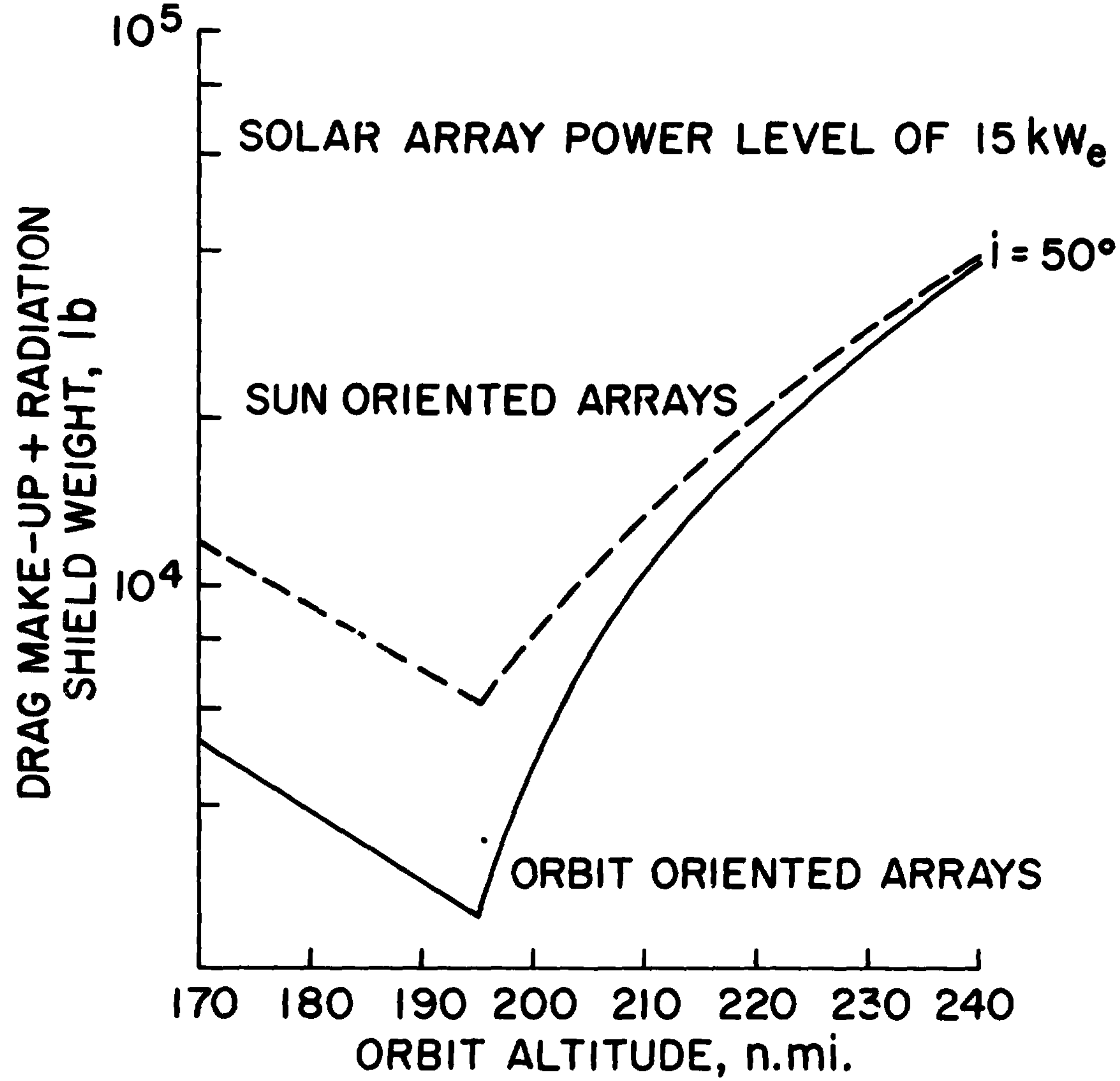


Figure 24.

DRAG MAKE-UP AND RADIATION SHIELD SYSTEMS WEIGHT FOR MISSIONS C AND D

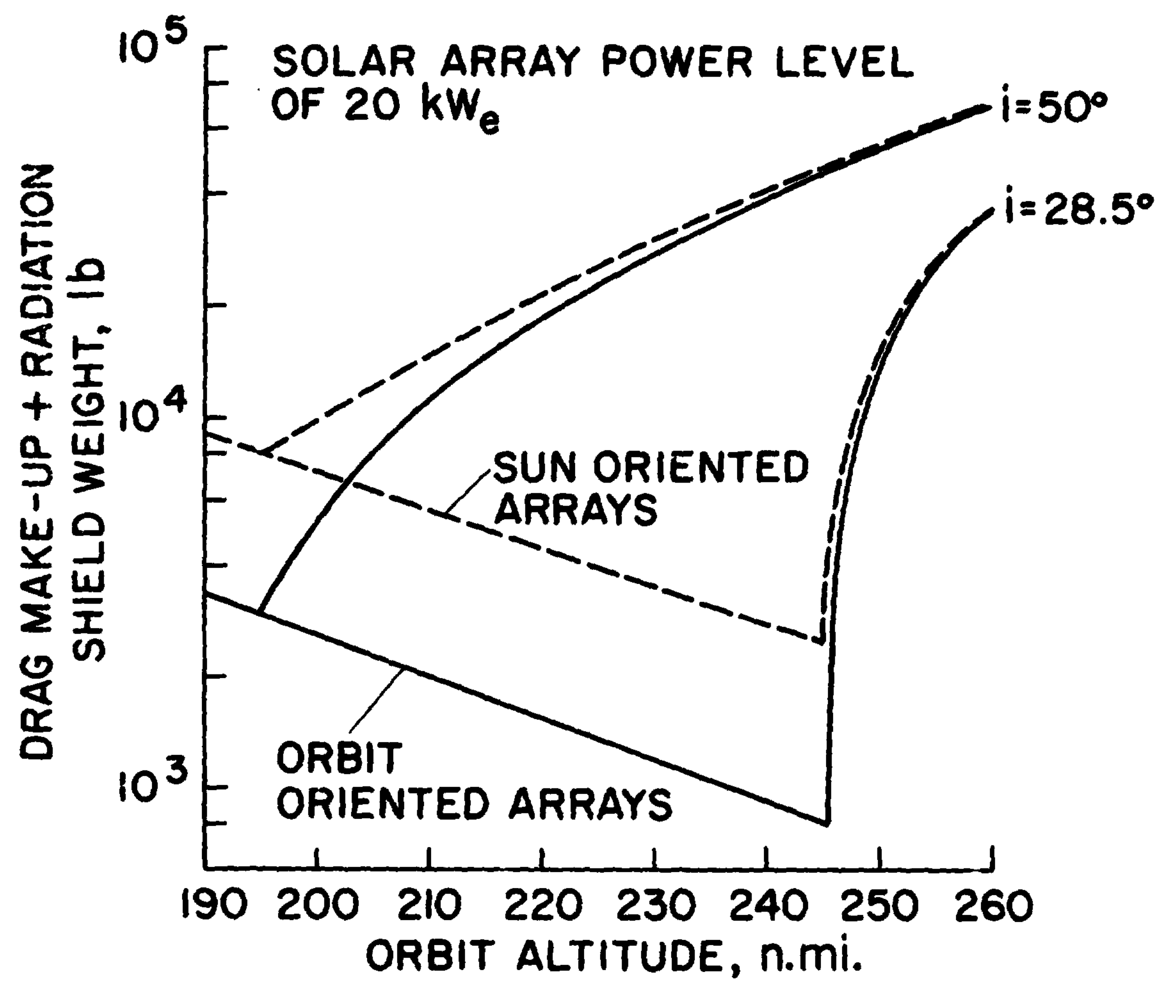


Figure 25.

were to be used to define the orbital characteristics, then the four missions' orbits would be as follows:

Mission	Inclination Degrees	Altitude nautical miles
A	28 1/2	245
B	50	195
C	28 1/2	245
D	28 1/2	245

Also of interest in addition to maintaining the space station in its desired altitude, is the rate at which each mission spacecrafts' orbit would decay after the conclusion of their missions. Figure 26 shows the orbit decay histories for each of the above mission orbits. For these curves, it was assumed that the array orientation was maintained and that the spacecraft was kept end-on to the direction of travel after the drag makeup system had been shut down. It can be seen that the orbits of missions A, C and D decay very slowly and nearly independently of which of the two array orientations is maintained. In fact at the altitude of 245 n.m., these spacecraft could be left for at least 18 months before there would be orbit decay sufficient to indicate imminent atmospheric reentry of the station. This should give mission planners some leeway in deciding on future spacecraft reuse.

It is also apparent from figure 26 that the orbit for mission B, with its initial altitude of only 195 n.m. decays very rapidly. In fact if its arrays are left sun-oriented, it decays to 130 n.m. altitude in only 140 days. If the arrays would be oriented parallel to the orbit path when the drag makeup system was shut down, then it would take 290 days to decay to 130 n.m. altitude. Post-program reactivation of mission B would be a more attractive alternative if the space station's altitude were to be increased before storing it in orbit. If this space station's altitude were increased by 25 n.m. to 220 n.m., then the station, with sun-oriented arrays, would take 360 days to decay to 130 n.m. and nearly 740 days if these arrays are stored orbit-oriented.

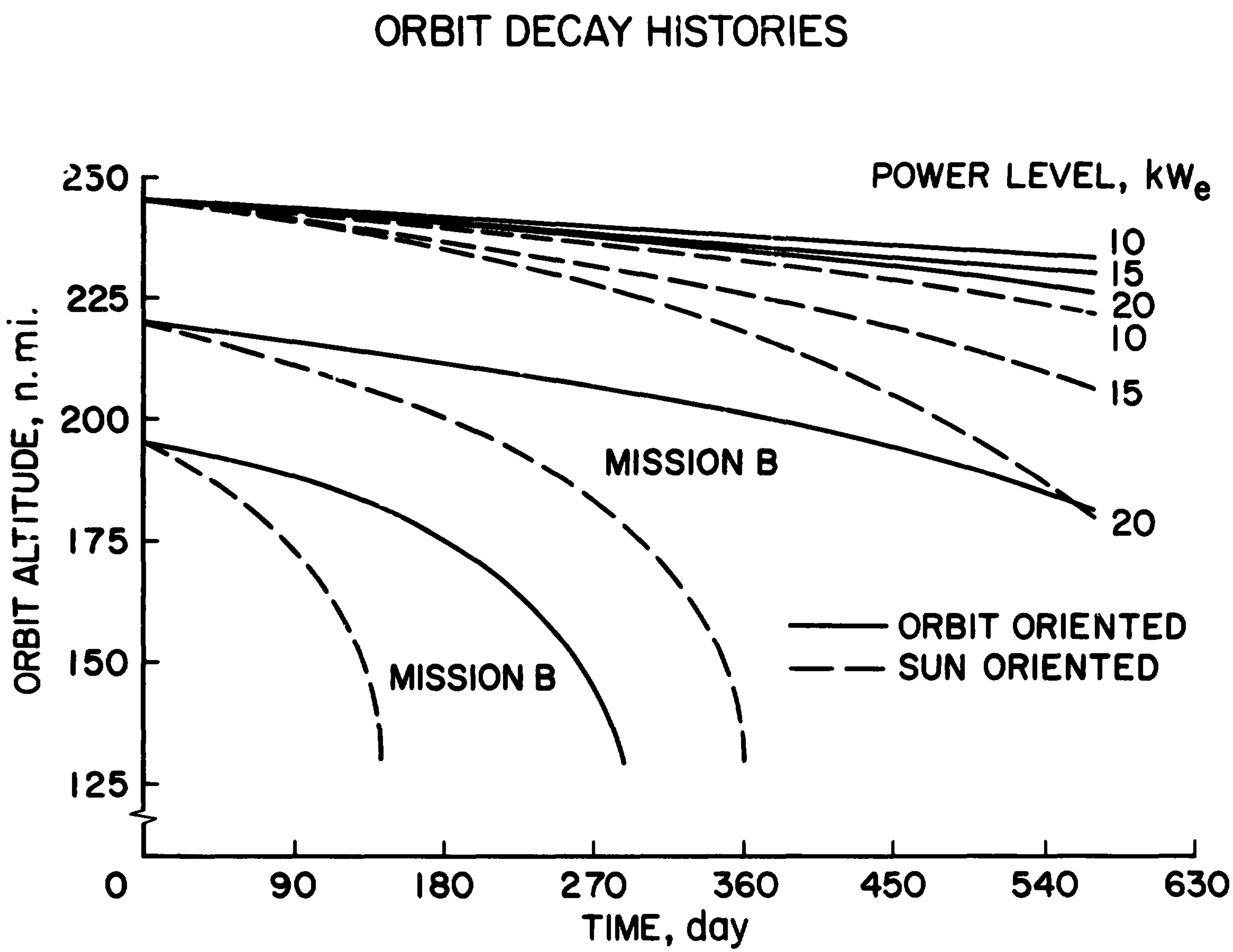

ORBIT DECAY HISTORIES
POWER LEVEL, kWe
10
15
20
10
15
20
MISSION B
MISSION B
ORBIT ORIENTED
SUN ORIENTED
ORBIT ALTITUDE, n.mi.
250
225
200
175
150
125
TIME, day
0
90
180
270
360
450
540
630

Figure 26.

Interim Space Station Weights

This section is more than just a statement of the weights for the several interim space stations. As was pointed out earlier, only those systems that may impact or be impacted by the mission requirements were evaluated in detail. This section integrates all the systems for each of the four missions with the requirements of these missions. When all the systems are considered, some systems influence the mission operational characteristics as much as the experimental program.

Except for the second mission which is largely devoted to experiments related to Earth observations, the mission requirements of the other mission experiments do not have specific orbit altitude nor inclination requirements. So as to determine their mission orbits, missions A, C, and D were examined and tradeoffs performed between the experiment needs, radiation and micrometeoroid shielding requirements, drag makeup fuel, logistic vehicle capacity, and launch vehicles capabilities. An orbit inclination of 28 1/2 degrees permits use of the full capability of the launch vehicle, and it was the selection. The orbital altitude at which the weight requirements for radiation and micrometeoroid shielding and for drag makeup fuel is the lowest, results in the lowest total weight spacecraft, and this altitude of 245 nautical miles was the selection. Mission B, the Earth observations mission, has its orbit inclination of 50° defined by the ground survey and coverage requirement. Its orbit altitude of 195 nautical miles, on the other hand, was defined by minimizing the weight of the spacecraft. Table 17 summarizes the orbital requirements for each of the missions.

Table 17
Mission Orbit Requirements

Mission	Altitude Nautical Miles	Inclination Degrees
A	245	28 1/2°
B	195	50°
C	245	28 1/2°
D	245	28 1/2°

The weights for the six versions of the interim space stations are summarized in Tables 18 and 19. Table 18 lists the fixed equipment weights for each major spacecraft module. The current design weights for the Skylab I vehicle also are shown for comparative purposes, Ref. 7. If the experiments are ignored, the fixed item weight for the various spacecraft are almost proportional to the crew sizes. On the other hand, for the expendable supplies shown in Table 19, the expendable weights are closely proportional to the total man-days for each mission. The expendable items were broken down into two categories, those which are necessary or desirable at initial activation of space station, and those that can be supplied during any logistic flight. This was necessary so as to know what items needed to be launched as part of each interim space station. The effects of recovering the water, Mission C, and recovering both water and oxygen, mission D, can be seen by comparing these mission weights with those for mission B. Mission B has a crew of only six men while missions C and D have nine men each; yet the potable water weight is 23,000 pounds less for mission C than for mission B. The oxygen load for mission D, which is mostly for emergency, is 16,000 pounds less than that required for mission B. This combined weight saving is about equivalent to that for one Apollo CSM logistic vehicle.

Table 18
Interim Space Station Spacecraft Fixed Equipment Weights

	Skylab I 3-Men	Mission A 3-Men	Mission B 6-Men	Mission C Water Recovery 9-Men	Mission D H_2O & O_2 Recovery 9-Men	Mission 'C' Water Recovery 8-Men	Mission 'D' H_2O & O_2 Recovery 8-Men
Payload Shroud (Jettisoned)	25,025	25,025	25,025	25,025	25,025	25,025	25,025
Multiple Docking Adapter	9,135	7,855	7,875	8,035	8,035	7,875	7,875
Basic Structure	4,120	4,120	4,120	4,120	4,120	4,120	4,120
Meteoroid Protection	175	870	890	915	915	890	890
Operational Equipment	1,930	2,065	2,065	2,200	2,200	2,065	2,065
Experiments	430	400	400	400	400	400	400
Experiment Support	2,480	400	400	400	400	400	400
Airlock Module	15,920	15,570	15,760	15,890	16,815	15,870	16,725
Basic Structure	3,470	3,470	3,470	3,470	3,470	3,470	3,470
Fixed Shroud	5,175	5,175	5,175	5,175	5,175	5,175	5,175
Meteoroid Protection	-	120	120	120	120	120	120
Environmental Control	2,525	2,650	2,770	2,850	3,775	2,830	3,685
Electrical System	1,925	2,000	2,000	2,000	2,000	2,000	2,000
Displays and Instrumentation	1,225	1,225	1,225	1,225	1,225	1,225	1,225
Communications	530	530	600	650	650	650	650
Experiments	220	200	200	200	200	200	200
Experiment Support	850	200	200	200	200	200	200
Instrument Unit	4,325	4,325	4,325	4,325	4,325	4,325	4,325
Orbital Workshop	35,225	58,430	72,040	89,965	79,855	85,540	75,400
Basic Structure	12,220	12,220	12,220	12,220	12,220	12,220	12,220
Meteoroid Protection	1,100	6,060	6,210	6,360	6,360	6,210	6,210
Interior Partitioning	4,965	4,965	7,900	10,840	10,840	9,050	9,050
Attitude Control System	1,570	1,570	1,600	1,600	1,600	1,600	1,600
Electrical Power System	4,980	15,000	15,000	15,000	15,000	13,500	13,500
Environmental Control	690	690	1,380	2,620	3,840	2,400	3,590
Crew Support	950	950	1,700	2,450	2,450	2,200	2,200
Furnishings	2,710	3,170	5,025	6,975	6,975	6,510	6,510
Communications	335	350	450	550	550	500	500
Station and Launch Control	3,625	3,625	3,625	3,625	3,625	3,625	3,625
Experiments	1,850	3,920	8,180	22,825	* 2,095	22,825	* 2,095
Experiment Support	230	2,610	5,450	1,600	11,000	1,600	11,000
Film Storage Vault	-	3,300	3,300	3,300	3,300	3,300	3,300
TOTALS	89,630	111,205	125,025	143,240	134,055	138,635	129,350

* Modules Launched with Logistic Resupply

FOLDOUT FRAME 1

Table 19
Interim Space Station Spacecraft Expendable Supplies Weights

	Skylab I 3-Men	Mission A 3-Men			Mission B 6-Men			Wat
		Initial	Resupply	Total	Initial	Resupply	Total	Initial
Multiple Docking Adapter	690							
Supplies	690							
Containers								
Airlock Module	25,690	15,965	32,555	48,520	19,285	43,060	62,345	22,645
Oxygen	4,930	3,390	6,770		4,340	9,770		5,330
Oxygen Tankage	16,000	8,900	18,100		11,170	25,130		13,540
Nitrogen	1,100	940	1,880		940	2,010		940
Nitrogen Tankage	3,500	2,700	5,400		2,700	5,700		2,700
Coolant and/or Water	110	90	270		90	300		90
Water Containers	50	45	135		45	150		45
Orbital Workshop	19,205	12,345	38,880	51,225	20,635	68,080	88,715	47,655
Film	605							-
Film Radiation Storage	3,300							-
Attitude Control Fuel (N_2)	1,350	900	2,700		900	3,000		900
Attitude Control Tankage	2,650	1,800	5,400		1,800	6,000		1,800
Drag Makeup/Artificial "g" Propulsion			2,400		1,600	5,100		8,100
Drag Makeup Tankage			240		160	510		1,200
Potable Water Supply	6,200	4,050	12,150		8,100	27,000		11,510
Potable Water Tankage	3,000	2,025	6,075		4,050	13,500		5,755
Food	1,680	1,080	3,240		2,160	7,200		3,240
Food Containers	420	270	810		540	1,800		810
Experiment Support		2,020	5,330		1,205	3,610		400
Experiment Support Packaging		200	535		120	360		40
Experiments in Modules								
Launch Vehicle-Artificial "g" Propulsion								13,900
TOTALS	45,585	28,310	71,435	99,745	39,920	111,140	151,060	70,300

FOLDOUT FRAME 2

Mission B 6-Men		Mission C Water Recovery 9-Men			Mission D H_2O & O_2 Recovery 9-Men			Mission 'C' Water Recovery 8-Men			Mission 'D' H_2O & O_2 Recovery 8-Men		
Resupply	Total	Initial	Resupply	Total	Initial	Resupply	Total	Initial	Resupply	Total	Initial	Resupply	Total
43,060	62,345	22,645	55,975	78,620	18,135	36,195	54,330	21,605	48,290	69,895	16,990	31,620	48,610
9,770		5,330	13,220		1,690			5,000	11,260		1,480		
25,130		13,540	33,860		5,230			12,830	28,870		4,700		
2,010		940	2,200		940	2,200		940	2,010		940	2,010	
5,700		2,700	6,200		2,700	6,200		2,700	5,700		2,700	5,700	
300		90	330		5,050	18,530		90	300		4,780	15,940	
150		45	165		2,525	9,265		45	150		2,390	7,970	
68,080	88,715	47,655	41,760	89,415	24,930	86,410	111,340	44,220	37,680	81,900	21,495	82,330	103,825
		-											
		-											
3,000		900	3,300		900	3,300		900	3,000		900	3,000	
6,000		1,800	6,600		1,800	6,600		1,800	6,000		1,800	6,000	
5,100		8,100	5,100			5,100		8,100	4,800			4,800	
510		1,200	510			510		1,200	480			480	
27,000		11,510			11,510			9,520			9,520		
13,500		5,755			5,755			4,760			4,760		
7,200		3,240	11,880		3,240	11,880		2,880	9,600		2,880	9,600	
1,800		810	2,970		810	2,970		720	2,400		720	2,400	
3,610		400	1,270		830	3,775		400	1,270		830	3,775	
360		40	130		85	380		40	130		85	380	
			10,000			51,895			10,000			51,895	
		13,900						13,900					
111,140	151,060	70,300	97,735	168,035	43,065	122,605	165,670	65,825	85,970	151,795	38,485	113,950	152,435

LAUNCH AND LOGISTIC VEHICLE CAPABILITY

Space Station Insertion

INT-21 Launch Vehicle - Each of the interim space stations will be launched by a two-stage Saturn V consisting of the S-IC and S-II stages. This launch system is sometimes referred to as the INT-21. Inasmuch as the external structure of each interim space station is in reality the S-IVB stage the overall appearance of the launch vehicle/station at launch is similar to the more familiar three stage Saturn V. Figure 27 shows the launch vehicle arrangement and lists the stage characteristics.

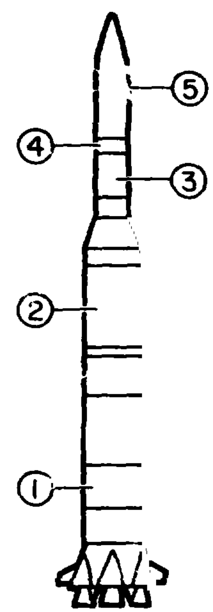
Since each of these vehicles will have been manufactured for the Apollo Program no major product improvements will be incorporated in their design. In particular the new, higher performance J-2-S engine will not be available for use aboard the S-II stage. Thus only one start of the S-II propulsion system is possible, dictating that the S-II stage firing be continuous from ignition through cut-off at insertion into the desired orbit. The performance envelope for such a continuous burn ascent is shown in Figure 28. The envelope was developed from a simplified computer program of launch vehicle performance. A particular value of performance for this launch vehicle from more extensive analysis in support of the Skylab I program is shown. Within the accuracy of other weight determinations for the interim space station, the accuracy for the performance envelope given is sufficient. The suggested orbit altitudes and inclinations for the interim stations, and the resulting gross payload capability to these orbits, are tabulated below:

Mission	Altitude (n.m.)	Inclination (deg)	Payload (lbs)
A	245	28 $\frac{1}{2}$	202,000
B	195	50	200,000
C	245	28 $\frac{1}{2}$	202,000
D	245	28 $\frac{1}{2}$	202,000

Although the use of Figure 28 is obvious, three considerations (other than the necessity for continuous burn) that influence the

LAUNCH VEHICLE CONFIGURATION

① FIRST STAGE (S-IC)		
	DIAMETER	33 ft
	HEIGHT	138 ft
	WEIGHT	5,030,715 lb FUELED
		2,287,850 lb DRY
	ENGINES	FIVE F-1
	PROPELLANTS	LIQUID [illegible]YGEN (3,308,605 lb, 340,531 gals.) RP-1 (KEROSENE) (1,428,855 lb, 209,792 gals.)
	THR[illegible]T	7,620,427 lb AT LIFTOFF
② SECOND STAGE (S-II)		
	DIAMETER	33 ft
	HEIGHT	81.5 ft
	WEIGHT	1,060,420 lb FUELED
		80,220 lb DRY
	ENGINES	FIVE J-2
	PROPELLANTS	LIQUID OXYGEN (820,710 lb, 86,208 gals.) LIQUID HYDROGEN (158,23[illegible] 282,543 gals.)
	THRUST	1,101,[illegible]O 1,161,414 lb
	INTERSTAGES	11,465



③ INTERIM SPACE STATION		
	DIAMETER	21.7 ft
	HEIGHT	58.3 ft
④ INSTRUMENT UNIT		
	DIAMETER	21.7 ft
	HEIGHT	3.0 ft
⑤ PAYLOAD SHROUD		
	DIAMETER	21.7 ft
	HEIGHT	54.0 ft

Figure 27.

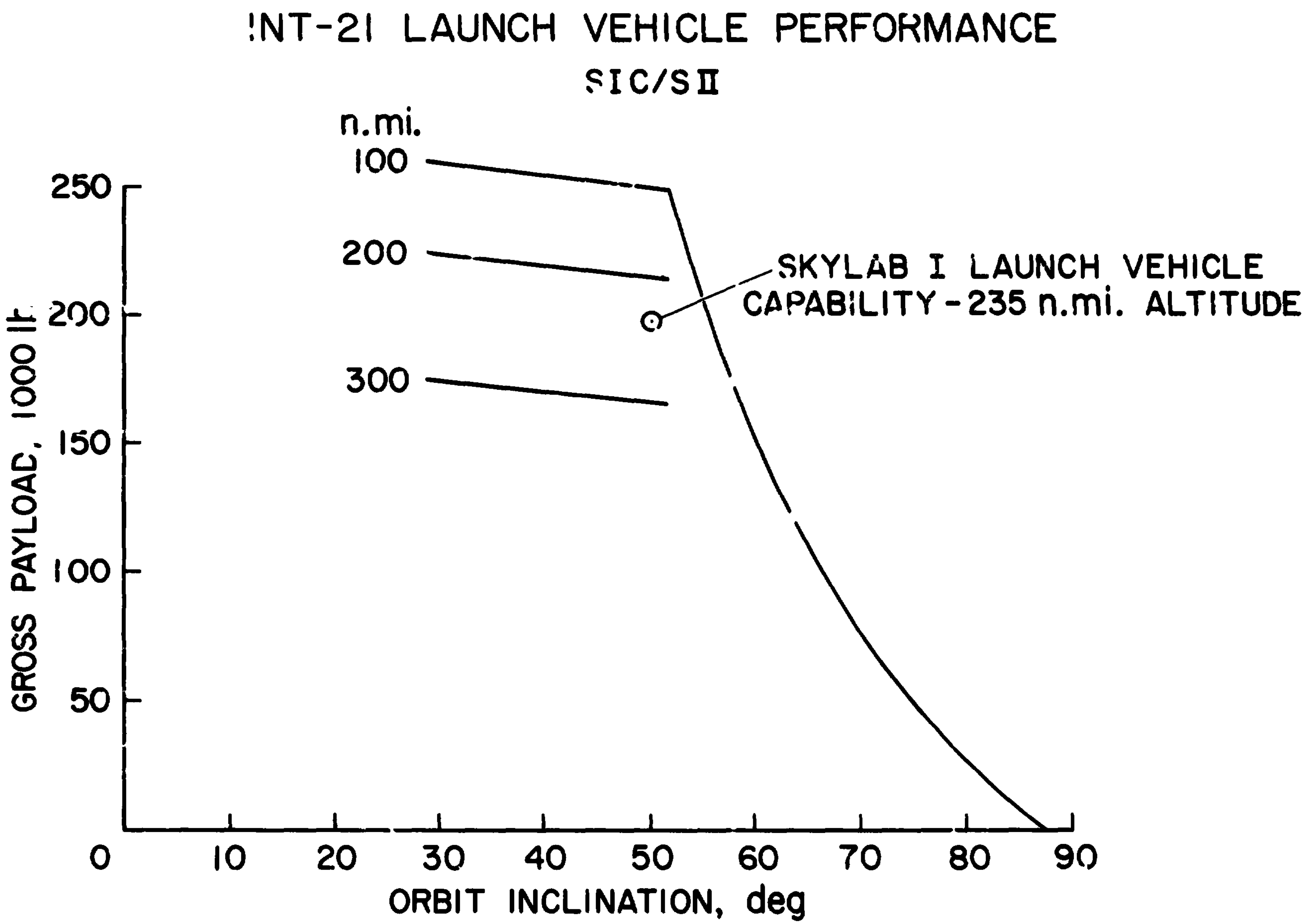

!NT-21 LAUNCH VEHICLE PERFORMANCE
SIC/SII
n.mi.
100
200
300
SKYLAB I LAUNCH VEHICLE
CAPABILITY - 235 n.mi. ALTITUDE
250
200
150
100
50
0
10
20
30
40
50
60
70
80
90
GROSS PAYLOAD, 1000 lb
ORBIT INCLINATION, deg

Figure 28.

allowable station weight need explanation: (1) the range safety constraints, (2) the location in the trajectory at which the payload shroud is jettisoned, and (3) the payload margin.

Range Safety - Recent information obtained from Saturn V launch operations analysis indicates that it should be possible to launch at azimuths as low as 45 degrees. The corresponding orbital inclination which results, predicated on a planar ascent trajectory, is 51.5 degrees. Higher inclinations would only be achieved by yawing the S-II stage, after orbital velocity has been attained, so as to rotate the orbital velocity vector. This latter assumption of a planar ascent is somewhat restrictive in that yaw steering toward the north would probably be performed during the ascent phase. This yaw steering maneuver would result in a slight gain in payload over that shown in Figure 28. Because of the potential hazard to the East coast of the United States and Canada, a detailed impact analysis is required to determine the limits to which this maneuver would be possible.

Shroud Jettison - The ascent trajectory concept currently under consideration for Skylab I calls for the payload shroud, weighing about 25,000 lbs., to be retained until orbit insertion. This concept was also assumed in the preparation of Figure 28. Thus the net weight allowable for an interim space station mission will be 25,000 lbs. less than indicated in the figure. A more standard ascent profile, of course, would call for the shroud to be jettisoned earlier in the trajectory with a resulting increase in net payload. If, for example, the shroud were jettisoned at S-II ignition the net payload would be lower than that shown in Figure 27 by only about 8000 lb. Since the type of payloads envisioned for the interim space station missions would need launch protection only against wind loads, the earlier shroud ejection and the 8000 pound payload penalty are more appropriate.

Payload Margin - As a point of reference the Saturn V payload capability for the Skylab I mission (h = 235 nm, i = 50°) is 197,000 lb, Ref. 7. The design weight of the total payload system (workshop, MDA, AM, IU, experiments and shroud) is about 165,000 lb. which indicates a payload margin of about 30,000 lb. Since most of the Skylab mission requirements are well established, it is apparent that this margin is intended to provide for growth in hardware weight. Thus it is reasonable to employ a similar margin of constant weight (about 30,000 lb) for the interim station missions rather than a margin based on a constant percentage of the launch vehicle payload capability. With these considerations for hardware growth and the descent altitude at which the nose shroud is jettisoned, the launch performance shown in Figure 28 would be reduced by 38,000 pounds, and the resultant values would be the maximum design weight for the mission spacecraft.

Logistic Vehicle

The Apollo command module and service module was the only vehicle considered to perform the logistic functions of carrying the crews to the space station and back to Earth. There were, however, two versions of the command module (CM) which were considered, the current 3-man version and a proposed 4-man version. Either version would land and be recoverable at sea using the current Apollo recovery operational methods. Some of the command module systems would of necessity be altered or modified to permit its quiescent storage in space for the 90-day crew rotation period. That equipment necessary for only the shorter lunar type missions would be removed to obtain storage volume and payload allowance for data and film to be returned to Earth. Those items that are removed just about equate in weight with those items which are required for the interim space station missions.

The four man version of the Apollo (CM) has been studied extensively by both industry and NASA for use as a logistic vehicle for Earth orbit missions. Adding the fourth man to the Apollo capsule does remove some of the practicality for using the crew seats as sleeping couches. The flights to an Earth orbiting space station would be less than 16 hours from time of start of crew's on pad checkout through lift off, injection, docking with the space station and the shutting down of the command module systems for its 90-day quiescent period. Return to Earth would be accomplished in less than this same period. Thus, there is no need to provide the convertibility of the crew seats to sleeping couches. The fourth crew man in the Apollo capsule would have the same type of landing shock attenuation built into his seat as is built into the present seats. Each man would have his own spacesuit and life support connections. A proposed six-man version of the Apollo command module also has been studied extensively. For this study, however, it was not considered for logistic support first, because the capsule would require extensive alterations and expensive changes, and second, the larger crew would not allow as much operational flexibility nor would it permit a more uniform number of men occupying the space station.

The Apollo service module (SM) serves as the cargo carrier, and as such requires the most extensive changes. The present Block II module has six compartments equally spaced around a center engine core. Four of these contain the tankage and fuel for the engine, and one contains the fuel cells and their reactants. For Earth orbit logistic support, the SM would act as a trans-stage and propel the Apollo CSM from the booster insertion altitude to docking with the space station. Upon Earth return the SM propulsion stage would bring the Apollo capsule to the lower orbit altitude at which the Apollo re-entry maneuver starts. This limited use of the propulsion system would require only one oxident and one fuel tank. The requirements for electrical power are also greatly reduced, and thus the equipment in the electrical power system bay would be greatly reduced. The reduction in electrical power needs reduces the amount of cooling radiator and consequent surface area coverage requirements.

Some of the fuel tanks used on the Apollo lunar module are more appropriate in size for the propulsion needs of the Earth orbital transportation maneuvers than those currently in the SM. If these are substituted for the existing fuel tanks, then the electrical power system components can also be installed in the same bays. These would fill the largest two of the six bays. This leaves four bays entirely available for storage of logistic cargo. This reconfigured service module has sometimes been referred to as a Block III version, Ref. 25. Two of the compartments have a volume of 175 cubic feet each. If the logistic cargo were all liquid, these four compartments could hold about 35,000 pounds of water exclusive of its tankage. If the cargo were a typical mixture of dry logistic support items, these four compartments could then contain about 20,000 pounds. There is adequate volume in these four SM bays to carry all experimental support items except those which measure over 3 1/2 feet in diameter and 12 feet in length. It must be understood that these cargo bays are only accessible through an extra vehicular activity; however the space traverse required is only slightly more than the conical height of Apollo, about 15 feet. The transfer of liquid cargo could be through built-in piping and it would require only the attachment of hose lines at the docking port. The bulk cargo transfer would be more difficult for it would require opening the compartment outer panel and transfer of items by crew EVA activity and some type of an endless line arrangement.

Table 20 summarizes the weights for the 3-man and the 4-man Apollo CSM spacecraft, and it includes the crew and the Block III service module without its logistic cargo load. Tables F-1 to F-4 in Appendix F contain detailed breakdowns of the weights given in this table. The effective weight of the launch escape system is reduced by about 6500 pounds when it is ejected soon after the second stage ignites.

Table 20
Apollo Logistic Vehicle Weight

Items	3-men	4-men
Fixed		
Command Module	11,145	12,035
Service Module	11,260	11,260
Launch Vehicle Adapter		
Saturn IV-B	4,155	4,155
Titan III-M	5,000	5,000
Launch Escape System (1)	2,700	2,700
Expendable		
Command Module	1,550	1,915
Service Module	5,930	5,100
Total		
With SIV-B Adapter	36,740	38,165
With Titan III-M Adapter	37,585	39,010

(1) Based on carrying LES until 29 seconds after second stage ignition (200 lbs) and yaw steering (700 lbs)

Logistic Launch Vehicles

Only three launch vehicles were considered for the logistic support for the interim space station missions. Each of them has currently developed systems, however as a launch vehicle, only one has had full flight qualification. These three vehicles are the Saturn I-B, the Titan III-M, and a third, having solid rocket motors as the first stage and a Saturn IV-B as the second stage. The latter vehicle has been named SRM-Saturn-IV-B for convenience. The characteristics and launch capability for each vehicle will be considered separately.

Saturn I-B - The Saturn I-B vehicle was developed to flight qualify the Apollo spacecraft and its systems. It consists of a Saturn I-B first stage and a Saturn IV-B second stage. In qualifying the Apollo spacecraft, the Saturn I-B was flight qualified for manned launches. The payload adapter is the same as is used on the Saturn V launches of the Apollo CSM spacecraft. Figure 29 shows the launch capability for the Saturn I-B at various inclinations and altitudes. It can place between 33.000 and 35,000 pounds into 250 nautical mile altitude orbits and at inclinations of greatest interest for space station operation.

Titan III-M - The Titan III-M was designed and configured for logistic support of the Air Force Manned Orbiting Laboratory, MOL program. The seven segment solid rocket motors for the Titan launch vehicle had just about completed their qualification tests when the MOL program was cancelled. The payload launch capability for this vehicle is given in figure 30. This vehicle can place about 25,000 pounds into a 250 nautical mile orbit. The Titan III-M was configured to be used with a Gemini 2-man vehicle, and thus it does not have as much payload capability as the Saturn I-B. To use the Titan III-M as a launch vehicle for the Apollo CSM, it will be necessary to construct an adapter section to mount the Apollo CSM on top the Titan. The vehicle would have a slight hammerhead shape at launch. There has been considerable experience with hammerhead configurations especially for the unmanned systems, and there should be no unexpected undesirable characteristics to prevent the qualification of this combination as a man rated launch vehicle.

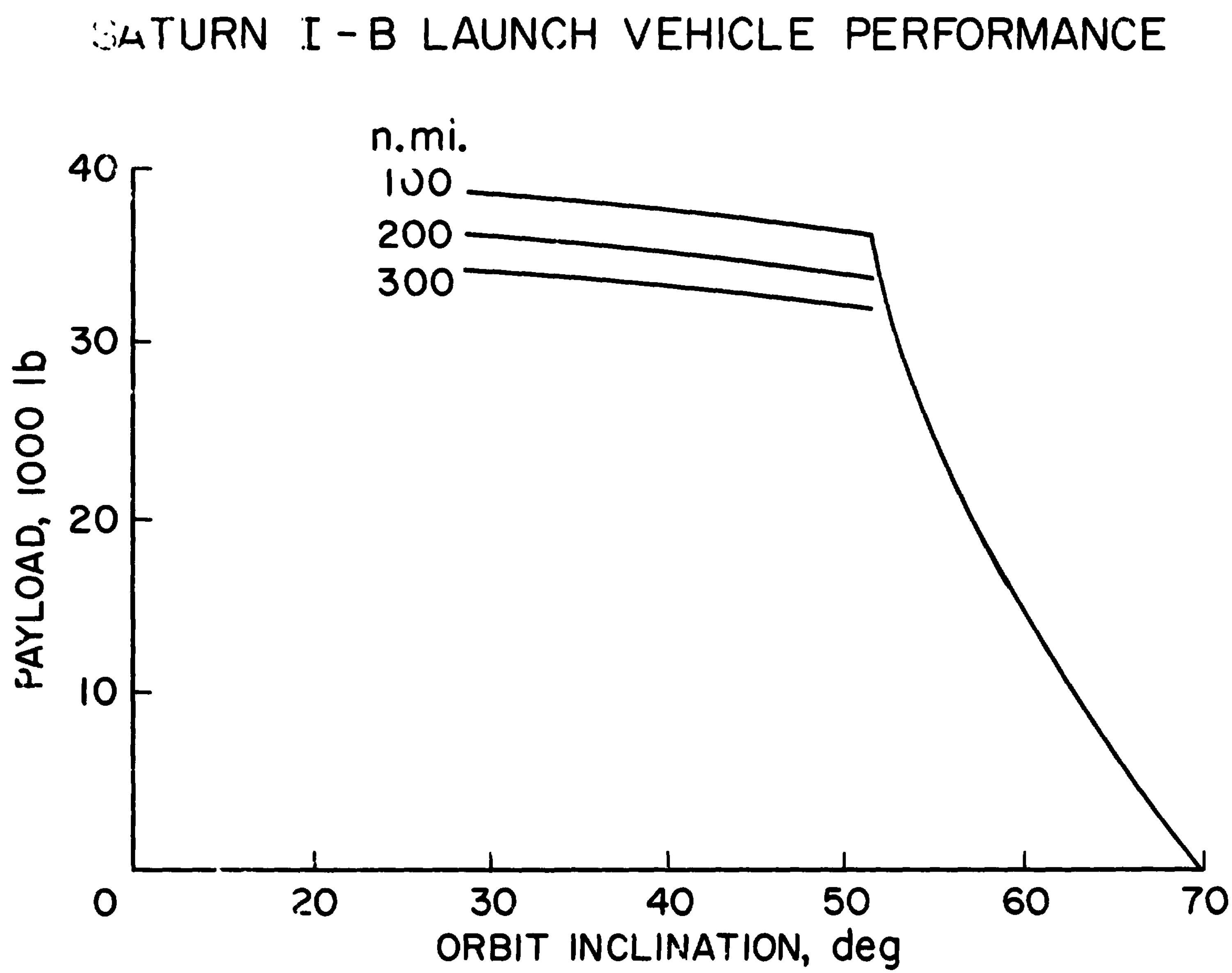

SATURN I-B LAUNCH VEHICLE PERFORMANCE
n.mi.
100
200
300
PAYLOAD, 1000 lb
40
30
20
10
0
20
30
40
50
60
70
ORBIT INCLINATION, deg

Figure 29.

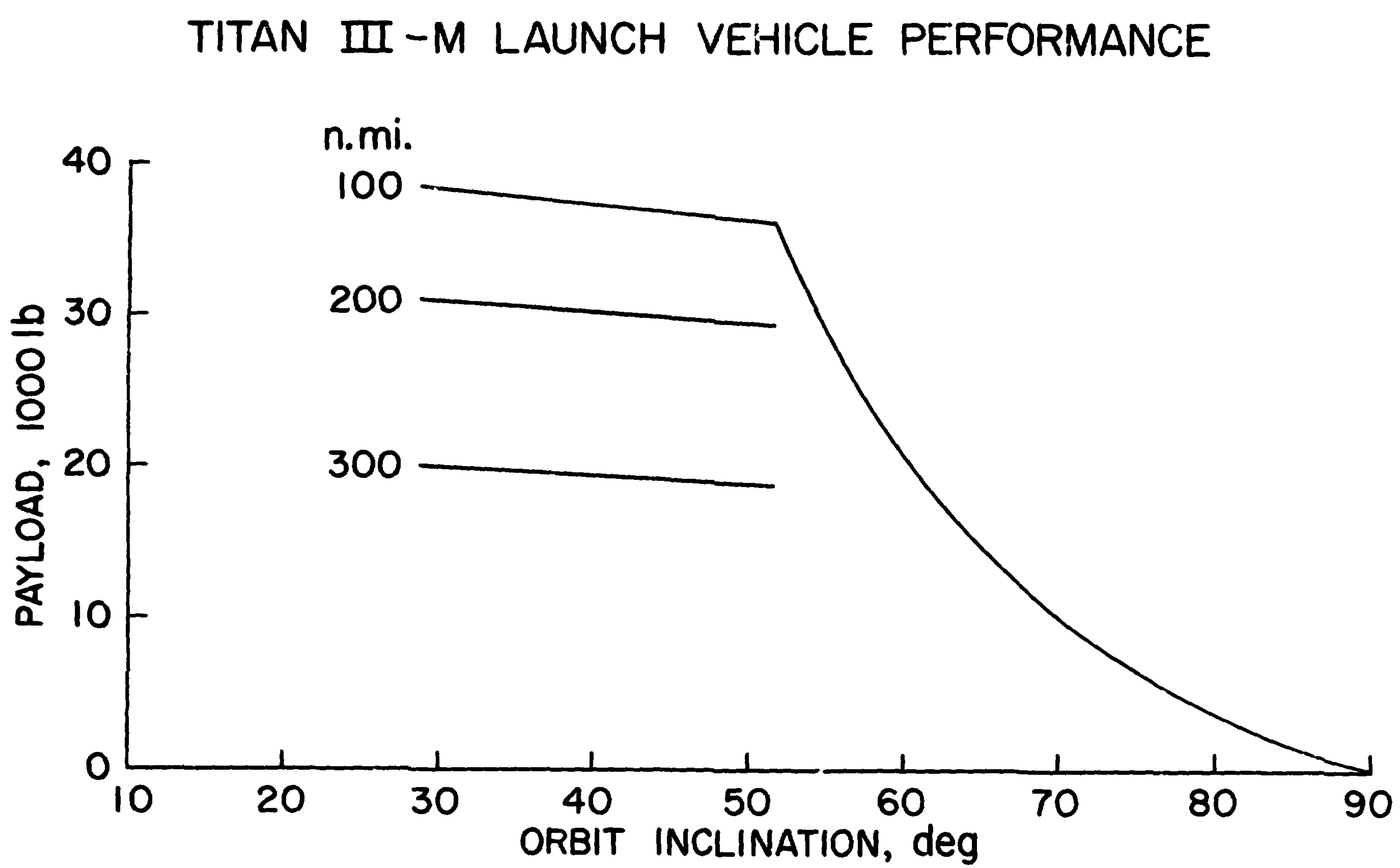

TITAN III-M LAUNCH VEHICLE PERFORMANCE
n.mi.
100
200
300
PAYLOAD, 1000 lb
0
10
20
30
40
ORBIT INCLINATION, deg
10
20
30
40
50
60
70
80
90

Figure 30.

SRM-Saturn IV-B - This vehicle has not been assembled, but its components have had extensive flight experience. Three 120-inch, seven segment, solid rocket motors, the same as those used with the Titan vehicles, comprise the first stage, and the SIV-B would be the second stage. Since the launch vehicle is a new configuration, the interconnections of the solid rockets as a first stage would need to be designed and tested, as well as the interstage with the SIV-B. However, the development of the vehicle should not be extensive for the solid rocket motors had almost completed their test phase under the MOL program. The design of the structural components and the clustering of the rocket motors should profit from this previous flight experience. To qualify this vehicle, it should require not many more tests. The calculated performance envelope for this launch vehicle is shown in Figure 31. This vehicle can place over 55,000 pounds of payload in a 250 nautical mile orbit at the inclinations of interest for this study.

Mission Accomplishment

The spacecraft weights and logistic support requirements for each mission have been detailed earlier. These payloads need to be compared with the launch vehicles' capabilities so as to determine the suitability and feasibility of each to support these studied missions. Table 21 summarizes for the six studied missions the fixed weights and the initial support requirements for each mission spacecraft. The INT-21 vehicle net launch capability is indicated. Except for mission C, INT-21 can insert the interim station, its fixed weight items and all of its initial manning needs into each desired mission orbit. For Mission C which has the unusually large loads in connection with the artificial gravity experiment. some selection would be needed between those items necessary to be available for the first crew at initial manning and those items which could be brought up on the first logistic missions. For this mission, which has more frequent logistic flights, some relaxation in the criteria of the amount of emergency life support supplies on board the interim station may be necessary. For instance, by changing the emergency life support supply criteria to that to be sufficient for a 60-day emergency, it would permit the transfer of about 10,000 pounds from the initial manning loads to resupply loads. What items to be included with each station launch is

SRM-SATURN IV-B LAUNCH VEHICLE PERFORMANCE
THREE-120 INCH, 7-SEGMENT SOLID ROCKET MOTORS

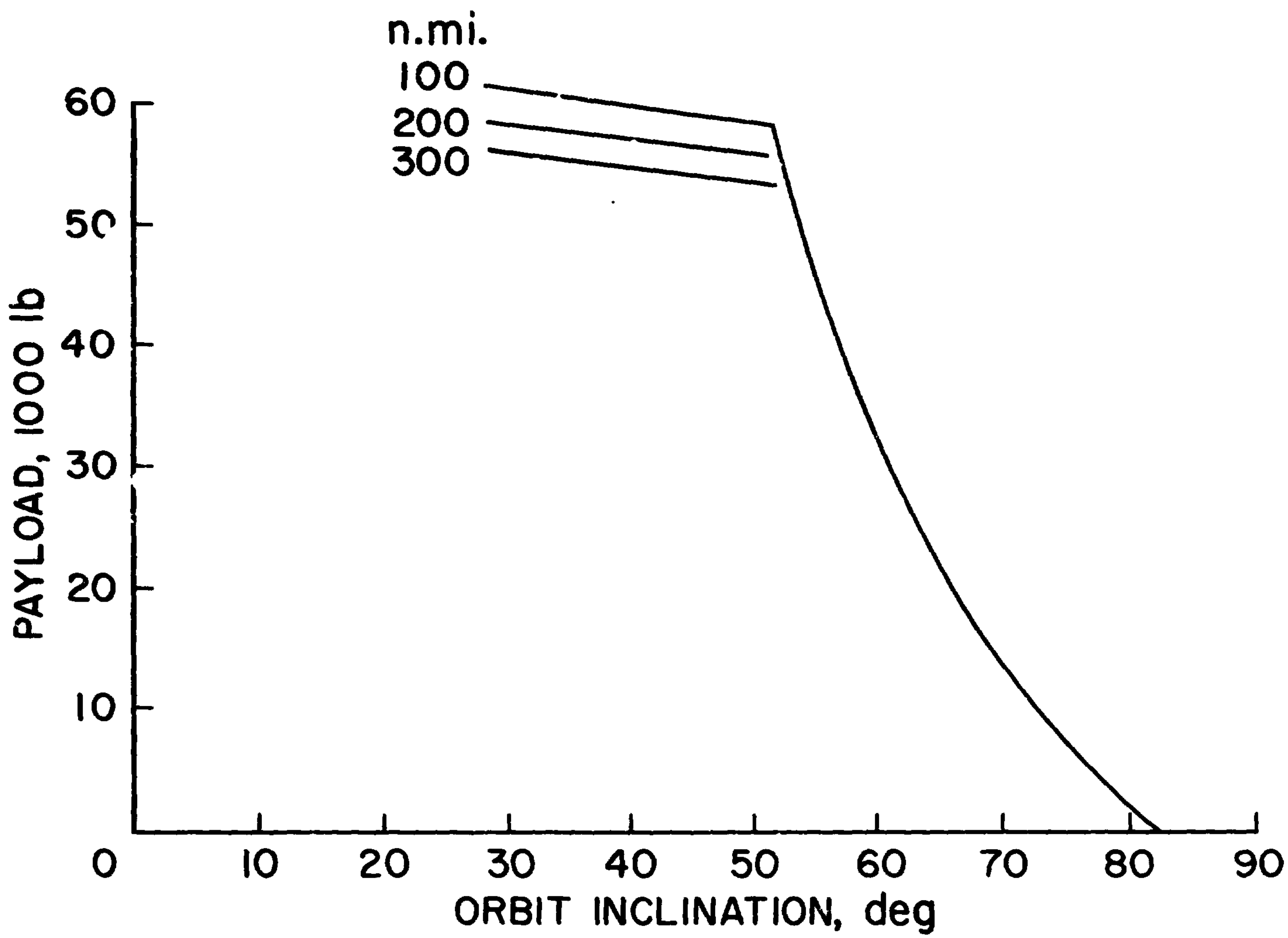


Figure 31.

Table 21
Interim Space Station Launch Requirements

	MISSION					
	A	B	C	D	Option C	Option D
Altitude, Nautical Miles	245	195	245	245	245	245
Inclination, Degrees	$28\frac{1}{2}$	50	$28\frac{1}{2}$	$28\frac{1}{2}$	$28\frac{1}{2}$	$28\frac{1}{2}$
Interim Space Station Weight						
Fixed Load*	94,180	108,000	126,215	117,030	121,610	112,325
Initial Manning Items	28,310	39,920	70,300	43,065	65 825	38,485
TOTAL	122,490	147,920	196,515	160,095	18[illegible],435	150,810
INT-21 Launch Vehicle						
Capability	202,000	212,000	202,000	202,000	202,000	202,000
Growth Allowance	30,000	30,000	30,000	30,000	30,000	30,000
NET	172,000	182,000	172,000	172,000	172,000	172,000
Launch Over-Capacity	49,510	34,080		11,905		21,190
Under-Capacity			24,515		15,435	

* Note: Includes 8,000 pound shroud allowance

of necessity outside the scope of a feasibility study. The important result shown in Table 21 is that each interim space station and a good percentage of its initially needed expendable items can be placed into the desired orbit for each mission.

Table 22 summarizes for the six missions the logistic loads and the launch capabilities for each of the three logistic launch vehicles. Those initial manning items not launched with the station for Mission C have been carried over to be included with its resupply loads. For the other missions, all of which have excess station launch capacity, some resupply items would be included with the initial manning items in order that the full utilization of each launch vehicle could be utilized. The resultant average resupply load for each logistic launch, ranges from about 5500 pounds for Mission A to 12,700 pounds for the optional 8-man version of Mission C.

Each logistic mission must launch an Apollo command module which carries the new crew to the station, and a service module which performs the docking maneuvers with the station. That launch capability of the logistic vehicle over that required for the Apollo CSM would be devoted to resupply items. Table 22 shows that neither the Saturn I-B nor the Titan III-M have load capability to carry any resupply items. In fact they cannot insert the Apollo CSM package as defined in this study into any of the mission orbits. The Titan III-M lacks over 10,000 pounds of capability to perform the logistic mission for just the Apollo CSM, and thus it should not be considered as a candidate vehicle for use with these missions.

The Saturn I-B lacks between 1500 and 3000 pounds of capability to launch this study Apollo CSM. This same Saturn I-B will be used to launch the 3-man Apollo CSM logistic vehicles in support of the Skylab I program. The logistic resupply load over and above the Apollo CSM for this Skylab I program is less than a hundred pounds for each launch. By careful staging, the Skylab I logistic missions can be performed by the Saturn I-B vehicle. Similarly, careful tailoring of the insertion maneuver and greater use of the service module propulsion system could probably increase the capability of the Saturn I-B so that it might insert the Apollo CSM into the interim

Table 22
Logistic Launch Vehicle Requirements

	MISSION					
	A	B	C	D	Option C	Option D
Altitude, Nautical Miles	245	195	245	245	245	245
Inclination, Degrees	28½	50	28½	28½	28½	28½
Number Logistic Launches	4	8	12	12	8	8
Apollo CSM Crew Size	3	3	3	3	4	4
Logistic Requirements						
Over or Under S.S. Launch Cap.	-49,510	-34,080	+24,515	-11,905	+15,435	-21,190
Resupply	71,435	111,140	97,735	122,605	85,970	113,950
NET	21,925	77,060	122,250	110,700	101,405	92,760
Logistic Load per Logistic Launch	5,480	9,635	10,190	9,225	12,675	11,595
Saturn I-B Launch Vehicle						
Capability	35,300	34,200	35,300	35,300	35,300	35,300
Apollo CSM Load	36,740	36,740	36,740	36,740	38,165	38,165
Net Logistic Payload	- 1,440	- 2,540	- 1,440	- 1,440	- 2,865	- 2,865
Titan III-M Launch Vehicle						
Capability	26,300	30,000	26,300	26,300	26,300	26,300
Apollo CSM Load	37,585	37,585	37,585	37,585	39,010	39,010
Net Logistic Payload	-11,285	- 7,5[illegible]	-11,285	-11,285	-12,710	-12,710
SRM-SIV-B Launch Vehicle						
Capability	57,300	55,700	57,300	57,300	57,300	57,300
Apollo CSM Load	36,740	36,740	36,740	36,740	38,165	38,165
Net Logistic Payload	20,560	18,960	20,560	20,5[illegible]0	19,135	19,135
Over Capacity per Launch	15,080	9,325	10,370	11,335	6,460	7,540

station mission orbits. However, this would result in the necessity to have the resupply items supplied by some other vehicle or means.

If unmanned logistic launches are to be considered, the Apollo service module would be necessary for each logistic flight so as to dock the cargo. The weight of the service module, its expendables, the launch vehicle adapter, and a nose cone would total about 24,000 pounds. This would leave for each unmanned Saturn I-B launch approximately 10,000 pounds for resupply items. This capability is about equal to the average resupply load necessary for each manned logistic mission. Thus, for the full logistic support of these missions by the Saturn I-B it would require one unmanned logistic resupply launch with each manned launch. And by this means, the Saturn I-B could be considered to be a candidate logistic launch vehicle for these missions. However before a final decision were to be made, a more detailed study would be required of Mission D to investigate how the Isotope Brayton power and the astronomy experiments modules could be handled operationally.

The third evaluated logistic launch vehicle, the SRM-SIV-B has an excess of resupply payload capability for each logistic launch. This excess varies from 7,600 to 15,000 pounds depending on the mission. This excess capability realized in the seven segment solid rocket motor, however causes one to wonder if the logistic loads could be satisfied with fewer solid motor segments. Although a five segment motor has not been tested for use with manned launches, it has been and is being used extensively for unmanned launches. Thus the capability of a three, five-segment solid rocket motors first stage with a S-IVB second stage was determined. Its calculated capability turned out to be about 10,000 pounds less than that for the seven segment SRM-SIV-B. As such, it would have capability to support only about half of the logistic missions. Because of this, the five segment vehicle could be a candidate launch vehicle, but it would not be as attractive. The seven segment SRM-SIV-B has excess capability which can permit either greater resupply loads or experiment hardware growth and this makes it the more desirable choice.

PROGRAM COSTS

Without manned spaceflight between the Apollo and the Space Station Programs, the operational competence built up at Kennedy in launch capability and at Houston in mission control can either be maintained on a sustaining basis or abandoned. It is felt that these facilities should be sustained, and thus this interim space station program evaluates the burden of these sustaining costs. This program also carries another burden, a characteristic of an interim program, and that is the costs which are required to reactivate vehicle manufacture and to develop new vehicle parts. This section discusses the program costs and the rationale used for developing the costs. It includes the program costs for several alternate methods of logistic support.

Costing Rationale

NASA in their manned spaceflight programs usually have divided the costs of its programs into three major areas. These are spacecraft or experiment development, spacecraft acquisition, and mission operations. In this study the same cost divisions were used.

The area of development costs takes several meanings depending on the past history of the item being considered. For example, the Saturn IV-B stage has been developed as a propulsion stage; however, except for Skylab I use, it has not been engineered nor tested as a manned mission module. Each of the four Saturn IV-B vehicles have unique requirements for each of the four missions and thus they are each assessed a cost for development. On the other hand for vehicles whose manufacture has stopped, such as the Saturn I-B, it has been assumed that the jigs, fixtures and tooling have been stored and that no new development would be necessary. However there are expenses that occur to reactivate manufacture, and these are treated the same as development costs.

In determining the amount of money that would be needed for developing various spacecraft and their components, it was necessary to synthesize results from studies, historical cost data, and estimates based on developing similar types of spacecraft (Ref. 26). This synthesis required a great amount of subjective judgment. Table 23 lists the vehicle development costs as developed. These costs for each vehicle have been assumed to be spread over a four-year period prior to the year of first launch. The development cost of $250 million for the interim space station includes the development and integration of the airlock module and the multiple docking adapter as well as the life support and crew accommodations. The development cost of the Apollo command module into a four-man capsule includes the cost of development of the current Block II service module into a Block III SM with a cargo storage and carrying capability as well as a propulsive stage. The start up costs for the 3-man Apollo CSM includes the development costs required for a Block III SM. The Titan III-M has not been man flight qualified; nor does it have an Apollo CSM launch vehicle adapter; nor are there the launch facilities for integration of logistic vehicles. These items are included in the cost for development for this vehicle. The components that are combined to form the SRM Saturn IV-B launch vehicle have all had extensive flight tests, and thus this vehicle's development costs include those costs entailed in the integration of the two stages, the manned flight qualifications and the launch operations of component integration and assembly.

Table 23
Vehicle Development Costs

Vehicle	Millions
Interim Space Station (DWS)	250
Apollo CSM, 3-men*	100
Apollo CSM, 4-men	225
Saturn I-b*	70
Saturn SIC-SII (INT-21)*	130
Saturn IV-B*	30
Titan III-M	250
Three 120" 7-Segment Solid Rocket Motor plus SIV-B Stage (SRM-SIV-B)	80

*Start-up

For accounting purposes, the development and acquisition of the experiments to be performed during the flight program have been assessed as development costs. The logic for this assumption is that each experiment and its equipment are unique, and that the equipment is built and developed concurrently. The estimates for the costs of these experiments were the most difficult to make. The prime source of experiment cost data was that available in the NASA "Experiment "Blue Book", Ref. 27. The other source was the results of some internal analysis of historical data and which related flight instrument costs to both size and weight of experimental equipment. One assumption regarding the experiments was that although an experiment was to be performed on subsequent flights, its equipment and technology would be altered sufficiently to require newly developed equipment for each interim space station flight. Except for the first mission whose launch date only permits a 4-year development period, the development costs for the experiments were spread over the 5-year period prior to the year of each space station's launch.

Most of the vehicles used in this program have been used in the Apollo program and thus the costs to acquire them are well documented. Table 24 lists the values as used in this study for the costs to build and deliver one vehicle. In these unit costs it has been assumed that there is continuity of production of vehicle type over the period of vehicle usage. Acquisition costs for each vehicle were assumed to be spread over the three year period prior to its launch year.

Table 24
Vehicle Acquisition Costs

Vehicle	Millions
DWS	150
Saturn I-B	50
INT-21	80
Titan III-M	25
SRM SIV-B	40
Apollo CSM, 3-men	45
Apollo CSM, 4-men	50

The determination of the program operational costs was perhaps the easiest to estimate, largely because in this area, there is the greatest amount of data. In the past there have been overlapping manned flight programs, and the only difficulty in analyzing their operational costs has been in determining what proportion of the launch and operational costs were being assessed to each program. The operational costs, in this study, have been assumed to include those services furnished in connection with the launch site assembly, checkout and launch of each spacecraft and those in support of the spacecraft while in orbit. It was assumed that there would be other manned operational launches before the start of this program and others after, but none during the four interim space flights. Thus this program would carry the burden continuously for the operation of Kennedy's launch complex and Houston's Mission Control during the time period of this program. This basis for assessing these costs might be considered to be arbitrary, however for evaluative purposes it does give good approximations to total program cost. If there are other program launches during the interim space station's flight period, then the operational launch costs would be shared, and these estimated costs for the interim program would be high. On the other hand, if there is a period with no launches either immediately before or after the interim program, the launch operation would need to be maintained during these periods and the interim space station program should be charged with this burden, and then in this event the program total costs would be low.

It was assumed that all the manned operational cost would be assessed to this program continuously over the period from the year of the first launch through the year that the last spacecraft is flying. This is a period of nine yars, and it annually averages over 500 million dollars. Table 25 lists the annual operational costs for each vehicle used in this program. The INT-21, since it is the major launch vehicle, is assessed with the major operational costs. If this vehicle were not being used, some of this operational burden for Kennedy and Houston would need to be carried by the other vehicles being used, and thus their operational costs of necessity would be higher than shown. The construction of launch facilities for the Titan III-M and the SRM-SIV-B launch vehicles are not included in these operational costs, for it has been assumed that these types of facilities normally would be charged to the NASA facility acquisition and not to a flight program.

Table 25
Annual Launch and Mission Control Costs

Vehicle	Millions
Saturn I-B	150
DWS	65
INT-21	350
Titan III-M	50
SRM - SIV B	70
Apollo CSM, 3-men	95
Apollo CSM, 4-men	100

Program and Option Costs

By integrating the development, operational, and vehicle procurement costs for the DWS, INT-21, experiments, 3 and 4-men Apollo spacecraft and the three candidate logistic launch vehicles, the total resources required for six options of this program were developed. Figure 32 shows the annual program rate of expenditures for the program for each of the three logistic launch vehicles supporting only the three-man Apollo. These costs are for a logistics launch total of 36 for each of the three launch vehicles, and Figure 32 thus shows the relative annual costs between each of these three launch vehicles. The SRM-SIV-B launch vehicle costs about 170 million more during the peak cost years than if the Saturn I-B vehicle were used and the use of the Titan III-M correspondingly would be about 150 million less than the Saturn I-B. One thing should be remembered in considering these annual costs, and especially during the peak years, and that is that the operational costs of about 600 million per year for the period from 1976 through 1984 have been included in these annual costs. Thus, the highest annual program expenditures shown are about 1 billion dollars over that required to support the annual operations costs.

Logistically, the Titan III-M vehicle cannot perform even the required manning functions in this program, and thus further consideration of it is

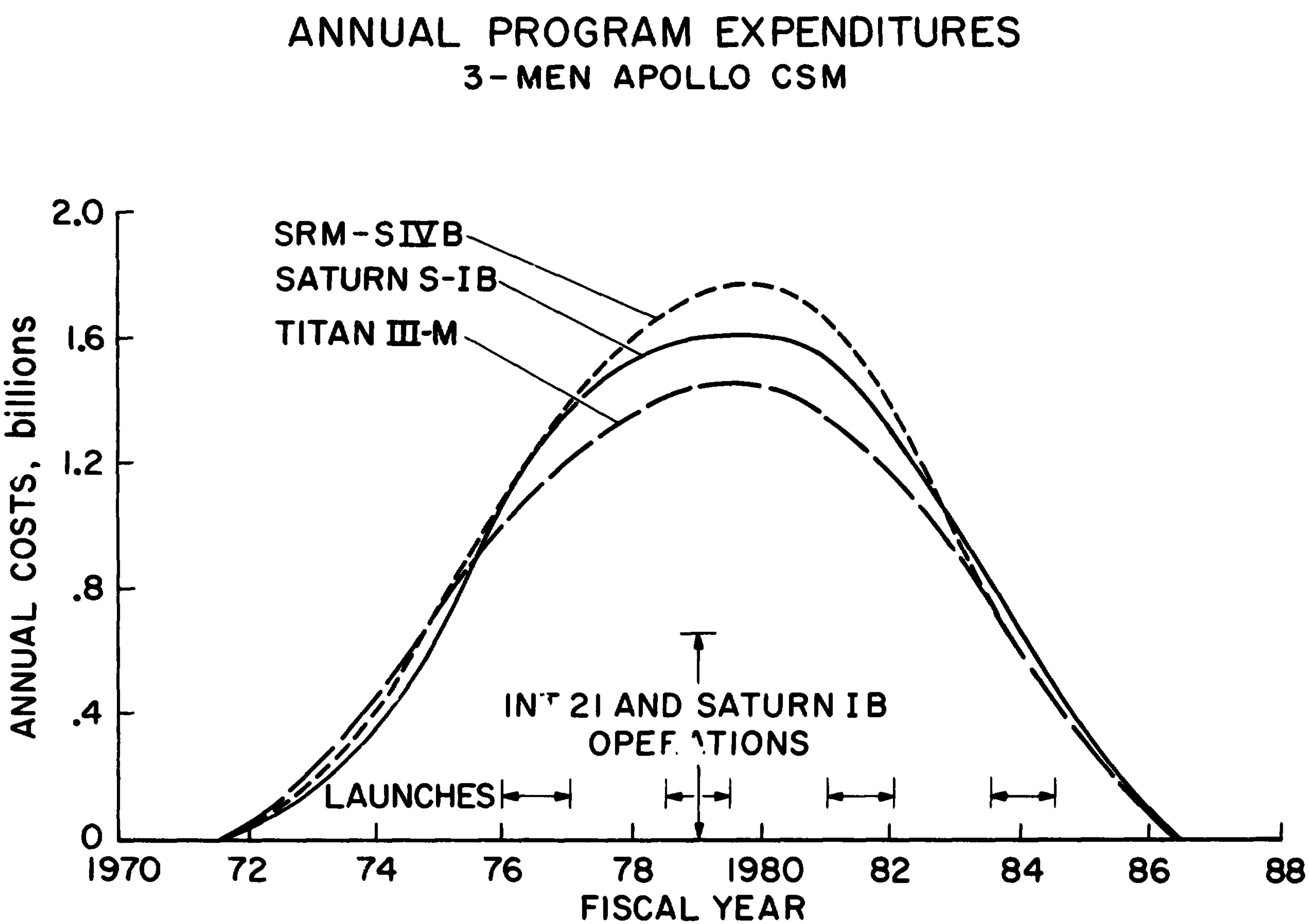

ANNUAL PROGRAM EXPENDITURES
3-MEN APOLLO CSM
SRM-SIVB
SATURN S-IB
TITAN III-M
INT 21 AND SATURN IB
OPERATIONS
LAUNCHES
ANNUAL COSTS, billions
2.0
1.6
1.2
.8
.4
0
1970
72
74
76
78
1980
82
84
86
88
FISCAL YEAR

Figure 32.

inappropriate. The Saturn I-B launch vehicle, by the addition of about an equal number of unmanned launches and orbit dockings to its manned launches, can satisfy the logistic requirements. Figure 33 shows the annual costs of the program with the full logistic requirements being performed by either the Saturn I-B or the SRM-SIV-B launch vehicle. These annual costs are for the optional program in which 3-men Apollo CSM's are used for Missions A and B and the 4-men versions are used for Missions C and D. These programs include costs for 28 manned launches for either launch vehicle and the additional required 29 unmanned logistics launches for the Saturn I-B vehicle. The fewer launches and vehicles required in the optional program brings the peak funding when the SRM SIV-B vehicle is used down to 1 1/2 billion per year. This value is 220 million less than the annual funding peak when the 3-men Apollo CSM's are used exclusively. The doubling of the number of logistic launches that are required when the Saturn I-B vehicle is used raises its annual program costs above that for the SRM-SIV-B costs and to an annual cost peak of 1.82 billion dollars.

The peak funding for any of the five considered options of the program occurs in 1979 and 1980. To determine the effect on the peak funding level by a change in launch interval, the costs were assessed for an increase in launch interval of the interim space stations from 2 1/2 years to 3 1/2 years. The longer launch interval reduces the peak level by about 200 million dollars, and it does delay the year when peak funding first occurs to 1981 or later. However, this stretchout in launch interval increases the total program costs by about 2 billion dollars; since three more years of operation costs have been added to the total program costs.

Table 26 summarizes the total costs for this program both for when only the 3-men Apollo CSM's are used, and for the optional program in which 4-men Apollo CSM's are used for the latter two missions. The costs shown in this table are those required to completely man and supply the four missions by either logistic or launch vehicles. The costs for additional unmanned logistic missions required by the Saturn I-B to satisfy the logistic needs of the missions are included in the vehicle acquisition and operations costs. The program costs for use of the Titan III-M vehicle are not given, for this vehicle cannot satisfy the crew requirements.

ANNUAL PROGRAM EXPENDITURES FOR COMPLETE LOGISTIC SUPPORT
3-AND 4-MEN APOLLO CSM'S

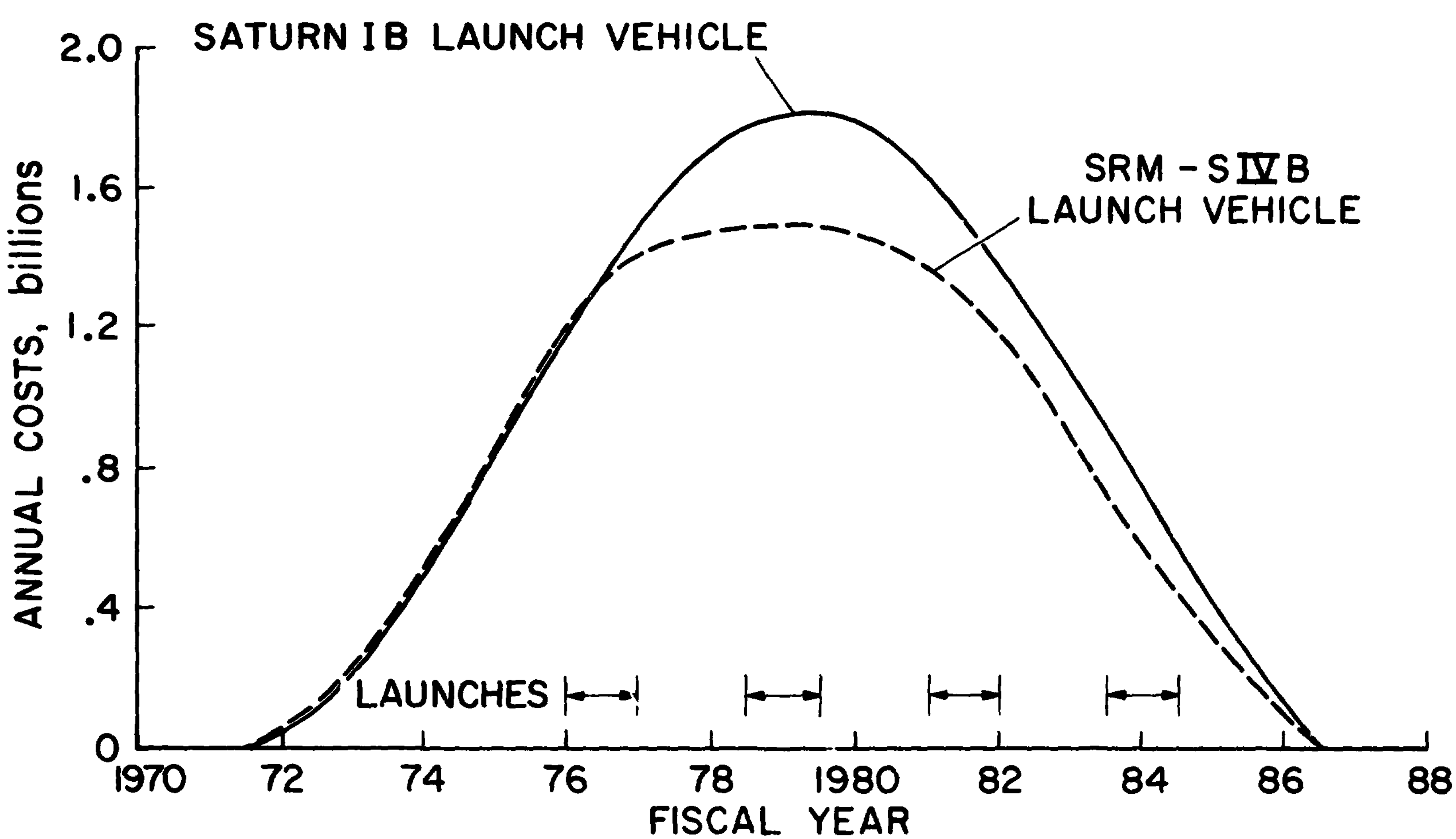


Figure 33.

Table 26
Program Costs - Billions of Dollars

3-Men Apollo CSM's

Logistic Launch Vehicle	Development	Experiments	Vehicle Acquisition	Operations	Total
Saturn I-B	1.170	1.970	6.730	5.940	15.810
SRM SIV-B	1.235	1.970	5.680	5.220	14.105
3 and 4-men Apollo CSM's					
Saturn I-B	1.395	1.970	5.770	5.965	15.000
SRM SIV-B	1.460	1.970	4.720	5.245	13.395

The SRM Saturn IV-B logistic launch vehicle, because of its excess payload capacity and thus requiring only the minimum number of vehicles to change the crew, has the minimum total program costs with either of the Apollo vehicles used. For this program's total cost of $14 billion, the launch operations and mission control contribute over $5 billion to this total. The costs for vehicle development and acquisition are almost $7 billion. These vehicle costs on a per mission basis are comparable to some of the costs being suggested for a second Skylab mission.

The need for additional Saturn I-B logistic vehicles increases its total program costs about $1.7 billion over that for the same program with SRM SIV-B vehicles. An interim type program should minimize costs and the development of new hardware items, and the program option which uses the SRM SIV-B vehicle is the most attractive. In terms of insurance against equipment and hardware weight growth it also is the best, for it has a minimum of 6500 pounds in excess capability available for each logistic launch. This excess capability could be used for such items as future hardware weight growth. The introduction of a 4-man Apollo CSM logistic vehicle into the program for the last two missions can produce a vehicle acquisition savings of 17 percent and a total program savings of $700 million. In spite of the need to develop the 4-man Apollo and the SRM SIV-B vehicles, this option is the least expensive of all the program options; and in spite of the reduced crew size, it does accomplish almost 90 percent of the desired scientific experiments.

CONCLUDING REMARKS

This study of an interim Earth orbital manned flight program gives the results for one feasible plan to maintain continuity in manned flights between Skylab I and the Space Station. These results represent just one possible approach of many alternatives available to NASA in its pursuit of manned space flight, and they could be used to help redirect the NASA programs if some unforeseen delay of the Space Station might develop.

From programmatic considerations, this four mission, interim space station program is a viable manned flight program. The detailed study showed that:

1. A scientific experiment program could be performed over the four missions, that would be equivalent to about two years of similar scientific effort on the Space Station.

2. Derivatives of the Skylab I spacecraft, ground fitted and supplied, could be the mission spacecraft.

3. Spacecraft systems such as electrical power and life support could be technically evolved, developed, and flight qualified from those as used on Skylab I to those necessary for the Space Station/Base program.

4. Either three-men Apollo CSM vehicles or combinations of three and four-men Apollos could perform the crew rotation functions. Only the SRM SIV-B launch vehicle in one launch has the capability to insert the manned Apollo spacecraft and the required resupply load into orbit. The Saturn I-B launch vehicle requires an unmanned logistic resupply launch for each launch of a manned Apollo CSM vehicle.

5. The program which uses the SRM SIV-B vehicle because it requires the least number of vehicles has the least cost.

6. The acquisition of mission spacecraft and experiment hardware would cost over $8 billion. All items used in the development of the interim space station program vehicles will not be directly applicable to the development of either the Space Station or Space Shuttle; and thus the monies spent in the interim space station program will not be entirely recoverable in either the development of the Space Station or the Space Shuttle.

The technology implications from this study were:

1. Serial flight developments of several types of water and oxygen recovery units are possible in this type of program. Any recovery of the water or oxygen by even the experimental units is worthwhile, for they can reduce large logistic loads.

2. Development of low weight solar panels and battery systems of 15 to 20 kilowatt capacity are highly desirable for Earth orbit spacecraft.

3. Self contained type nuclear power systems could be flight tested, operated and evaluated in this program.

4. Frequent logistic missions permit timely supply of repair items. Repairability and maintainability of spacecraft components should be a design criterion.

5. Development of stability systems that would maintain the attitude and control of the mission spacecraft beyond its initial mission period, would make the reactivation of these spacecraft possible.

REFERENCES

1. Candidate Experiment Program for Manned Space Stations, NASA Office of Manned Space Flight, September 15, 1969.

2. Space Station Program Phase B Definition Study - Experiment Support Requirements Analysis, Vol. I, NASA Contract NAS8-25140, McDonnell Douglas Report Number MDC-G0605, July 1970.

3. Space Station Program Phase B Definition Study - Special Emphasis Summary, Experiments, NASA Contract NAS9-9953, North American Rockwell Report Number PDS-210, February 20, 1970.

4. Artificial Gravity Experiment, Medical-Physiological Protocol, Second Draft, NASA, March 27, 1970.

5. Skylab I Cluster Requirements Specifications, Revision 25, NASA, George C. Marshall Space Flight Center, RS003M00003, August 25, 1970.

6. Skylab I Program, Technical Summary, NASA-OMSF, August 1970.

7. Skylab I Weight and Performance Report, NASA, OMSF, SE016-001-1, August 1970.

8. Apollo Applications Program Specification, NASA, OMSF, SE-140-001-1, August 15, 1969.

9. Mission Requirements, Apollo Applications Missions AAP-1, 2, 3, and 4, NASA, George C. Marshall Space Flight Center, 1-MRD-001B, March 15, 1970.

10. Parametric Study of Manned Life Support Systems, NASA Contract NAS2-4443, McDonnell Douglas Report DAC-56712-56715, Vols. I-IV, January 1969.

11. Trade-off Study and Conceptual Designs of Regenerative Advanced Integrated Life Support Systems, NASA Contract NAS1-7905, Hamilton Standard Division of UAC, July 1969.

12. Space Station Program Phase B Definition Study 7th Technical Review, MDC G0592, June 1970.

13. Skylab I Systems Engineering and Program Definition Study Plan, George C. Marshall Space Flight Center, 6 May 1970.

14. Design Study Specifications for the Earth Resources Technology Satellite. ERTS A and B. Goddard Space Flight Center S701-P-3, Revised October 1969.

15. K. E. Peltzer: Apollo Unified S-Band System, IEEE 1965 International Space Electronics Symposium, pp 4B 1-12.

16. R. E. Spearing: Manned Flight Engineering Division, Goddard Space Flight Center, private communication, October 1970.

17. Space Station Program Phase B Definition Study, Vol. II, Mission Analysis, NAS8-25140, McDonnell Douglas Astronautics Company, Report Number MDC G0618, July 1970.

18. Parametric Study of Logistics-Type Entry Vehicles, Final Report, V3, Book 1, NAS2-2461 (Mission Analysis Division/Ames Research Center) with Missile and Space Systems Division of the Douglas Aircraft Company, September 1965.

19. Snyder, Joseph W.: Radiation Hazard to Man from Solar Proton Events, J. Spacecraft Vol. 4, No. 6.

20. Parametric Study of Manned Life Support Systems, Final Report, V2, NAS2-4443 (Mission Analysis Division - OART, Moffett Field) with McDonnell Douglas Astronautics Company - Western Division, January 1969.

21. Savin, R. C.: Sensitivity of Long-Duration Manned Spacecraft Design to Environmental Uncertainties, ASME Meeting, Los Angeles, California, June 16-19, 1968.

22. Lauderbach, P. W. Lt., USAF: An Analysis of Low Orbital Drag Constraints of Orbit - and Sun-Oriented, Solar-Cell Arrays, Paper presented at the 1967 Intersociety Energy Conversion Engineering Conversion Conference, August 13-17, 1967, Miami Beach, Florida.

23. Flight Performance Manual for Orbital Operations, Report NOR. 61-208, Contract Number 8-861 (Marshall Space Flight Center) with Northrop Corporation Norair Division, September, 1961.

24. Sterne, T. E.: An Introduction to Celestial Mechanics, Interscience Publishers, Inc., 1960.

25. Apollo Logistics Systems Status Review, Space Division of North American Rockwell Corp., Report Number PD 68-24, May 1968.

26. NASA's Manned Space Flight Program, Fiscal Year 1970, NASA Office of Manned Space Flight, April 29, 1969.

27. Experiment Program for Manned Space Stations, NASA Office of Manned Space Flight, May 1, 1969.

APPENDIX A
EXPERIMENT DESCRIPTIONS

The experiments planned for the four interim stations are summarized on the following pages. Since some of the experiments are used in several missions, they are referred to here by title and FPE number rather than by mission. This material was extracted from the "Green Book", "Experiment Support Requirements Analysis, Space Station Program Definition", NASA Contract NAS8-25140, McDonnell Douglas Astronautics Company, West, January 13, 1970.

Stellar Astronomy Module (FPE 5.2)

1) Description and Objectives

The objective of this FPE is to make stellar observations at higher resolution and in wavelength regions that are impossible to achieve from the ground. The objects that will be viewed include individual faint stars, galaxies and stellar clusters. Photometry of globular clusters, hot O and B stars and Cepheid variables will be used to determine the brightness and distances to those objects. High resolution spectroscopy will enable abundance determinations to be made of stars, comets and planetary atmospheres. The use of imaging photographic and electronic devices will produce high resolution pictures of star clusters, planetary surtaces and the structure of the Milky Way nucleus. Measurements of polarization will reveal gas and dust clouds in the spiral arms of the galaxy, and give information on their composition, density and movement.

The instrument used in this FPE is a 3 meter Cassegranian telescope. This telescope is a precursor for a true diffraction limited 3 meter telescope for future missions.

2) Cost and Availability

Cost - 130 million
Schedule - 1977-1978

Solar Astronomy Module (FPE 5.3)

1) Description and Objectives

This FPE will conduct visible, UV, and X-ray studies of solar granular structures and areas of high solar activity with higher spatial and spectral resolution than are achieved in the ATM instruments. The instruments in general are larger versions of the types used in ATM.

A tentative instrument complement will consist of a 1.5 meter aperture photoheliograph, a .25-.5 meter aperture spectroheliograph and spectrometer, two coronagraphs covering the 1-6 and 5-30 solar radii ranges and a 0.5 meter aperture X-ray grazing incidence telescope for both direct imaging and spectrometry of solar features. The instruments will be capable of receiving updated detectors over their lifetime.

2) Cost and Availability

Cost - 125 million
Schedule - 1977-1979

<u>Ultraviolet Stellar Astronomy Survey</u> (FPE 5.4)

1) Description and Objectives

The objective of this FPE is to photograph objective grating stellar spectra in the 1000 to 2000 °A ranges and develop instrument technology and in-flight procedures for large manned orbiting telescopes. Extension of photographic spectroscopy into the Lyman series of atomic and molecular hydrogen is a primary goal.

The observations are carried out with an all reflecting Schmidt telescope mounted on an ATM type stabilized platform. The excellent imagery and wide field of view of the telescope make it suitable for survey work in the far ultraviolet. The 0.3 meter telescope has all reflective optics and focuses the image onto an image converter that records the star field on a roll film camera operating in the visible part of the spectrum.

2) Cost and Availability

Cost - 5 million
Schedule - 1976-78

Space Physics Airlock Experiments (FPE 5.6)

1) Description and Objectives

This functional program element is comprised of a group of experiments whose primary goals are:

- To obtain data on the space environment in near earth orbit.
- To determine the effects and constituents of an induced atmosphere above the space station and to measure its temporal changes.

The following experiments are included in this FPE:

- S063 - Ultraviolet Airglow Horizon Photography
- S073 - Gegenschein/Zodiacal Light

Four other experiments have been defined by the "Blue Book" as part of this FPE. However, these experiments have common objectives with FPE's 5.17 and 5.18. For the purposes of this document the requirements for the following experiments will be included in the sections indicated.

- S149 - Micrometeorite Collection - included in FPE 5.18.
- T025 - Coronagraph Contamination Experiments - included in FPE 5.17.
- T027 - Contamination Measurements - included in FPE 5.17.
- T030 - Environmental Composition - included in FPE 5.17.

2) Cost and Availability

Cost - 1 million

Schedule - 1975

Cosmic Ray Physics Laboratory (FPE 5.8)

1) Description and Objectives

The cosmic ray physics laboratory is primarily an astrophysical observatory for high energy particles. The parameters of interest are flux of electrons, isotopic composition, energy spectra, and flux directionality.

The secondary objective involves the investigation of nuclear interactions using the cosmic rays as a source of particles. Also, the spallation cross-sections and products of higher atomic number particles can be determined over a wide energy range. The transverse momentum distribution will also be measured.

2) Cost and Availability

Cost - 50 million

Schedule - 1975

Earth Surveys (FPE 5.11)

1) Description and Objectives

This FPE comprises 19 sensors, ranging from the ultraviolet through visible and infrared to microwave. The purpose is to operate groups of these sensors simultaneously, conducting experiments in 1) Agriculture/Forestry/Geography, 2) Geology/Mineralogy, 3) Hydrology/ Water Resources, 4) Oceanography, and 5) Meteorology. Using the instruments commonly associated with earth resources and meteorology measurements, patterns and spectral signatures are formed which allow recognition of surface and atmospheric features.

Early earth resource sensors will not operate continuously, but will be run in an experimental fashion during several passes each day ranging up to 6 minutes duration. Pointing at truth sites will be an important part of the data taken to correlate signatures with known ground features. Meteorology sensors will also be of an experimental nature, but more nearly continuous operation will be required to obtain more nearly global atmospheric data.

2) Cost and Availability

The cost of the earth survey sensors is $40 million. Availability 1975.

Biomedical Research (FPE 5.13)

1) Description and Objectives

The overall objectives of the medical experiments program are described in two categories. The first is oriented toward the support and enhancement of man and his abilities in manned space flight. The second is oriented toward the advancement of medical science by making available to the medical community and its researchers of the opportunity to use the peculiar environmental factors of space flight in basic applied research.

The objectives for this study will be met by means of individual measurements to explore each of the nine areas of body function including: (1) Neurophysiology, (2) Cardiovascular Function, (3) Pulmonary Function and Energy Metabolism, (4) Nutrition and Musculoskeletal Functions, (5) Endocrinology, (6) Hematology and Immunology, (7) Microbiology, (8) Pathophysiology, and (9) Toxicology.

The laboratory system will consist of IMBLMS and the peripheral equipment including a Manned Onboard Centrifuge. The instrumentation will be arranged in modules which can be assembled into working consoles according to the requirements of the spacecraft and the medical experiments program for each particular mission. The pathophysiology and toxicology experiments will require test subjects consisting of small animals. These experiments will require Space Biology laboratory and technical support.

It is desirable that the medical experiments be conducted in an atmosphere as closely approaching that of the earth as the design of the spacecraft will permit.

2) Cost and Availability

The combined cost of IMBLMS, peripheral equipment (including the Manned Onboard Centrifuge) - 72 M.

Availability - Early 1976.

Man-System Integration (FPE 5.14)

1) Description and Objectives

The goal of the Man-Systems Integration Functional Program element is to achieve optimum utilization and support of man in advanced space systems. The broad objective is to determine the optimum uses of man's capabilities in space missions and includes the development of the techniques, technology, and equipment required for man to perform independently or in cooperation with ground personnel as a decision maker; a systems manager, operator and maintainer; and as a scientific investigator. Specific objectives are:

a) Quantify human capabilities for performing physical and mental work as an operator and maintainer of space systems and equipment, and as a scientific investigator, and to provide data for decisions on the appropriate man/machine mix.

b) Develop methods for crew selection, proficiency assessment, maintenance of skills, and to identify training requirements.

c) Determine man's individual behavior characteristics and group dynamics in space.

d) Develop operator equipment and technology for crew and cargo transfer, assembly, and maintenance internal and external to the space vehicle.

e) Develop the technology for habitable living areas for space vehicles. The Aerospace Medicine Facility will support the Man-Systems Integration FPE. This facility will include IMBLMS and peripheral equipment which includes an ergometer, airlock, a manned on-board centrifuge, and an acoustically isolated work area with a controlled light source.

2) Cost and Availability

The cost of the experimental equipment (excluding Aerospace Medicine Facility equipment) is estimated to be 10 million. The equipment required to support the Manned-Systems Integration FPE will be availabile by late 1975 or early 1976.

Life Support and Protective Systems (FPE 5.15)

1) Description and Objectives

The objective of LS/PS technology is to provide a controlled and physiologically acceptable environment for flight crews during all phases of a space mission. The life support system must, therefore, provide a pressurized shirtsleeve environment that also allows for pressure suit operation during normal or emergency conditions. It must supply food, water, and oxygen, provide for personal hygiene, and remove waste and contaminants. Lastly, the system must provide a thermal balance through utilization of available energy and dispersion of any excess heat. A basic assumption is that data from other completed space programs will be available for future utilization. This FPE is being included to provide critical information with respect to the environmental requirements, the design criteria for LS/PS, and the technology which will allow and assist men to perform effectively on future space missions. The much needed information will be provided through the following goals:

a) Investigation of the basic chemical and physical-phenomena, and their occurrence and rate of occurrence in those gravity-sensitive elements of future LS/PS components and subsystem assemblies.

b) Evaluation of advanced component, subsystem, and system performance, reliability, verification, and fit in the space environment.

c) Investigation of man-system and system-vehicle interfaces and demonstration of man's ability to accomplish maintenance and repair operations.

2) Cost and Availability

The combined cost of providing one conceptual experimenter unit for each of 14 different experiments described in Table II of FPE 5.15 is estimated to be 40 million dollars. The equipment required to support the Life Support and Protective System will be available by 1975.

Materials, Science and Processing (FPE 5.16)

1) Description and Objectives

The objective of this experiment is to establish the feasibility of processing and manufacturing products in space which can best be made in a near zero-gravity or extremely clean vacuum environment. The final products must meet a real and significant need of science and industry, and have a value exceeding the cost of space processing and transportation. In addition, the experiment must demonstrate man's capability to repair and operate equipment in space.

The experiments chosen for the program include:

Thin film - for higher purity and quality electronic devices.

Glass Casting - for large perfectly spherical lenses.

Spherical Casting - for precision spherical casting of minimum mass.

Composit Casting - for high strength composites.

Variable Density Casting - for high quality and uniformity of formed material.

2) Cost and Availability

Cost - $30 M

Availability - 30-36 months

<u>MSF Engineering and Operations</u> (FPE 5.24)

1) Description and Objectives

This experiment is a multi-facet experiment program aimed at developing engineering methods and operations concepts for future advanced missions. This FPE consists of a set of experiments that includes experiment hangar; guidance, stability and control technology; advanced power systems; advanced orbital EVA system[illegible]aintenance and repair techniques; logistics and resupply; manned [illegible]ncy and space living facilities; wireless power; and laser communi[illegible].

2) Cost and Availability

This is a Multi-Purpose Program, and availability and cost are complex functions of the final space station development activity. The gross cost is $400 million, and the equipment will be available during the 1975-1985 period.

Artificial-G Experiment (FTE 5.x)

1) Description and Objectives

This experiment provides operational and engineering data which may be useful in the assessment of artificial-g effects on man's ability to perform in space. This information impacts on the design of the artificial-g portion of the Space Base as to the allocation and configuration of experiments and support functions between the zero and artificial-g portions. Performance of the experiment requires that the Space Station be rotated at a rate and radius which is nominally equivalent to that expected for the Space Base. This requires that rotation-peculiar hardware and operations be employed, to the extent practicable, which will be evaluated during the experiment as a secondary and supporting objective. This may verify or impact the design and control of the Space Base artificial-g hardware.

APPENDIX B

ELECTRICAL POWER SYSTEM

This appendix gives more detailed data on the interim space station power requirements and system selection. Table B-1 shows the results from several Earth orbit spacecraft studies, the power requirements developed and the systems as defined. These results were used to evaluate alternate electrical power systems for the interim space station.

NASA has to date maintained parallel technical development on a number of systems that could supply electrical power from that required for the unmanned planetary spacecraft to the Space Station and eventually the Space Base. Each of the systems is an arrangement of several subsystems. The two primary subsystems are the energy source and the power conversion; however each system also includes power conditioning and power distribution subsystems. Table B-2 shows the major electrical systems which are under development. These are categorized by energy source and power conversion, and an indication is given for each of their approximate operating power output range.

Solar cells are technologically available with the 2400 square feet of panels scheduled to fly on Skylab I. This solar array can produce over 20 kilowatts in direct sunlight and it is configured to supply kilowatts continuously in Earth orbit, with its battery backup. Fuel cells are also technically ready, having flown on Gemini and Apollo. Of the nuclear systems the isotope Brayton system appears to be the most advanced technically, and it is expected to be operational first, then followed by the SNAP-8 reactor systems. One NASA estimate of the power capability and availability is given in Figure B-1, Ref. B-1.

Characteristic weights and areas for various candidate nuclear power source systems have been estimated by Lewis Research Center, Ref.B-2 and these are given in Table B-3 for [illegible]us operating conditions. Note the wide range of system we[illegible] equivalent power output of the nuclear reactor systems [illegible]s largely from variations in spacecraft arrangements, [illegible] assumptions, and reactor location.

Table B-1

TYPICAL EARTH ORBIT POWER REQUIREMENTS (WATTS)

Title	Single Launch Space Station	SNAP-8 Thermoelectric for Space Station	Large Space Station Power Systems	MORL Study	AAP Capability
Crew Size	6-9	6-9	9	6-9	3
Study Contractor	Boeing	NAR (AI)	TRW	Douglas	NASA
Reference	B-3.	B-4.	B-5.	B-6.	B-7.
Study Date	Oct. 1967	Sept. 1969	Oct.1968	Jan. 1966	Aug. 1969
Lighting		1400	1300	1063-268	OWS 1300
Instrumentation	2000	1000-3000		760-311	AM 1000
Communication and Data	560-1100		1900	1650-787	MDA 200
					CSM 1100
Stability and Attitude Control	690		600	609-181	ATM 2650
Crew Systems	670				
Life Support	1000	5000-7000	9000	3251-2735	
Thermal/ Environment		1200			
Experiments	1000-4000	2000	5700	3000	
Maintenance Check-out, Repair		1000			
Airlock/MDA		1000-2000			
CSM (Quiescent) each		1000-2000		3148-827	
Docked Remote Modules		1000-3000			
Subtotal	5400-9000	15600-26100	18500	13481-8409	6250
Contingency	-	1600-2600	1500	1300	930
TOTAL	5400-9000 (avg) (peak)	17200-28700 (low) (high)	20000 (nom)	14781-9709 (peak) (avg)	7180

Table B-2

CANDIDATE POWER SYSTEMS FOR SPACE STATIONS

	Energy Sources			
	Nuclear Reactor (SNAP-8)	Radio-isotope	Sun	Chemical
Thermoelectric	up to 25 KW_e			
Brayton	up to 100 KW_e	up to $\approx$ 15 KW_e		
Mercury Rankine	35-50 KW_e (SNAP-8 Sys)			
Solar Cells			up to 50 KW_e	
Fuel Cells				up to $\approx$ 20 KW_e

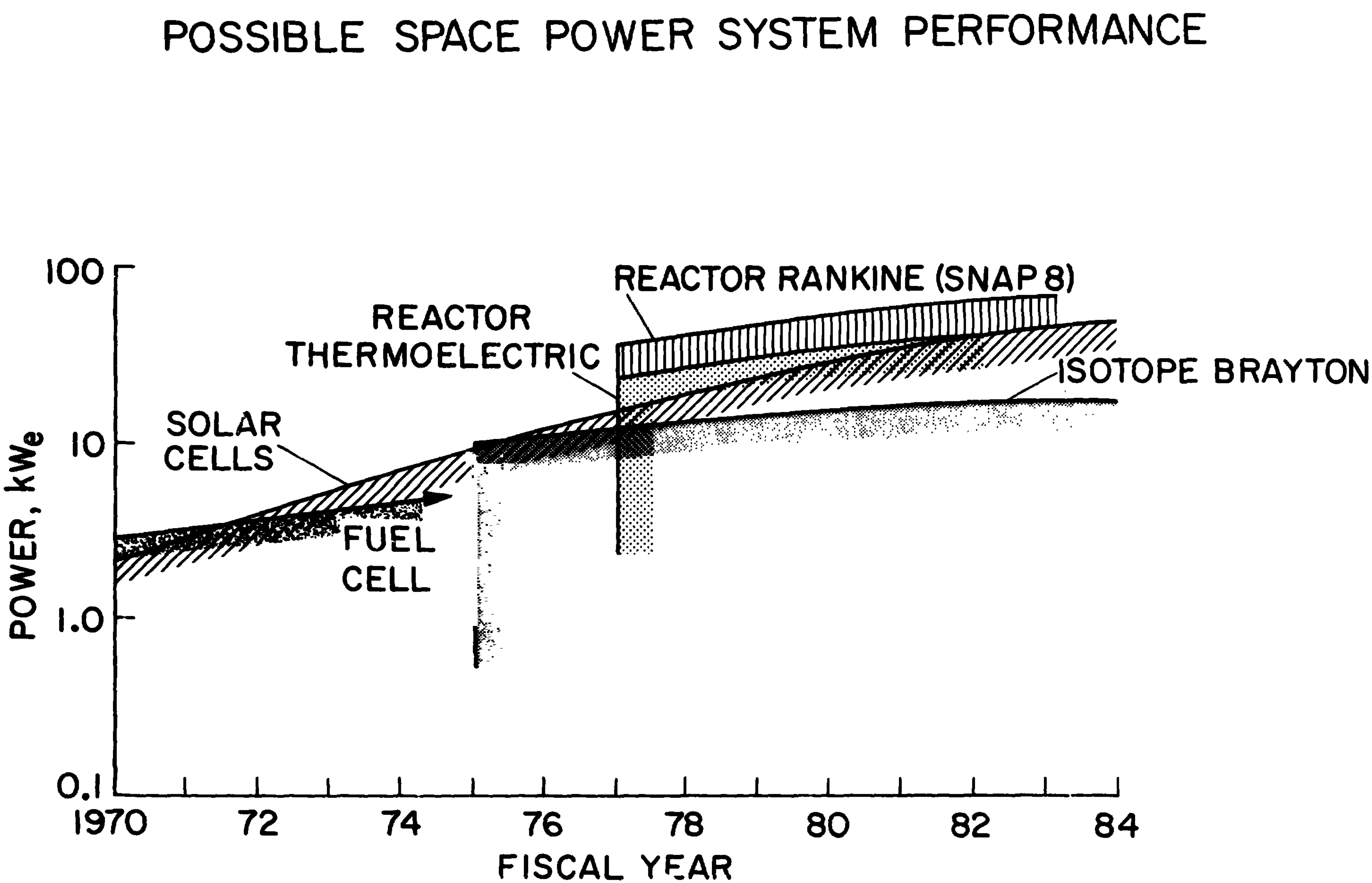

POSSIBLE SPACE POWER SYSTEM PERFORMANCE
REACTOR RANKINE (SNAP 8)
REACTOR THERMOELECTRIC
ISOTOPE BRAYTON
SOLAR CELLS
FUEL CELL
POWER, kWe
100
10
1.0
0.1
1970
72
74
76
78
80
82
84
FISCAL YEAR

Figure B-1.

Table B-3

NUCLEAR SOURCE POWER SYSTEM CHARACTERISTICS

	Reactor Condition	Electric Power KW_e	System Weight (Shielded) lbs	Radiator Area 450°R Sink Ft^2
Mercury Rankine	1300° F 350 KW_t	25	50,000 - 130,000	1150
Mercury Rankine	1300° F 600 KW_t	55		1850
Mercury Rankine	1100° F 600 KW_t "Benign"	25		2000
Thermoelectric	1300° F 600 KW_t	25		1900
Thermoelectric	1100° F 600 KW_t "Benign"	25		3200
Brayton	1300° F 110 KW_t	25		1600
Brayton	1300° F 600 KW_t	105		6500
Brayton	1100° F 600 KW_t "Benign"	100		10000
Isotope Brayton	100 KW_t	25	13,000	1500
Solar Cells		25	22,000	Array 7400 Area

The fuel cell being a chemical system, its weight is more than just a function of power level. For each kilowatt-hour of energy supplied, it reouires a total of about one pound of hydrogen and oxygen reactants. The fuel cell system weight, then, is sensitive to mission duration and/or resupply period. Figure B-2 shows how fuel cell weight varies as a function of resupply period and power output. Weight for other candidate systems is also shown for comparison. The technology readiness date for all systems is assumed to be 1975. The graph shows that mission or resupply periods of less than 30 days are required before the fuel cells look better than the other systems on only a weight basis.

POWER SYSTEM WEIGHT VARIATIONS

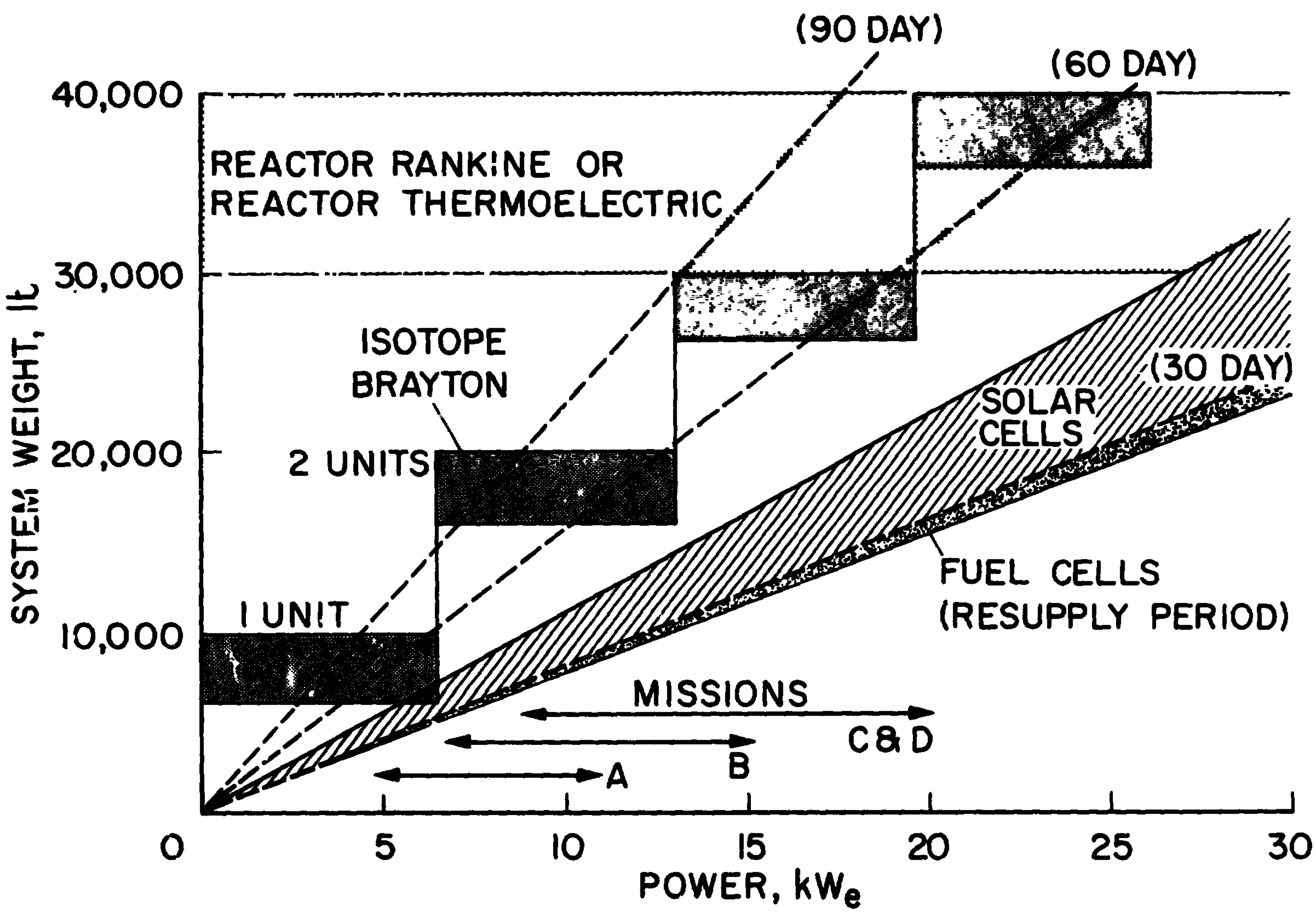


Figure B-2.

REFERENCES - APPENDIX B

B-1. Internal communication from NASA Office of Advanced Research and Technology.

B-2. Information on Nuclear Power Systems, Volume I, Internal Memorandum, NASA, Lewis Research Center, October 1969.

B-3. Saturn V Single Launch Space Station and Observatory Facility, The Boeing Company, NASA Contract NAS9-6816, Document D2-113536-1, November 1967.

B-4. Gylfe, J. D., et al, North American Rockwell, "25 KWe Reactor - Thermoelectric Power System for Manned Orbiting Space Stations", Proceedings of 4th IECEC, September 1969.

B-5. Boretz, [illegible] E., TRW, "Large Space Station Power Systems", October 1968.

B-6. Manned Orbital Research Laboratory (MORL) System Concept; Douglas Aircraft Company, Inc., NASA Contract No. NAS1-3612, Report No. SM-46089, January 1966.

B-7. AAP Capability, NASA Marshall Space Flight Center, August 1969.

APPENDIX C

PCM/PM TELEMETRY DATA TRANSMISSION LINK

Table C-1 tabulates those elements which contribute to or effect the telemetry downlink performance. In addition there is given the electronic characteristics of the telemetry equipment and facilities as used. It has been postulated that the high data rate link telemetry transmitter and antenna on the DWS should be comparable to those employed aboard the CSM for high data rate telemetry. The electronic characteristics given in Table C-1 are for a PCM/PM link. A PCM/FM link could have been evaluated, but its results would have been similar and it would have very comparable performance. The PM transmitter capability is 11.5 watts at high power, and was used in this calculation. Its low power rating is 3.24 watts. The spacecraft S-band high-gain antenna was assumed to be used in the wide beamwidth mode for maximum gain and coverage. The MSFN ground network consists of 30 and 85 feet diameter antenna insta lations, both of which would be used when available; however, for this analysis only the 30 foot disk is considered. The 30 foot antenna has better low angle coverage, and it provides greater acquisition range and time over a site than does the 85 foot antenna. However, because of its smaller gain, the 30 foot dish in the link is the condition to verify.

The free space loss is based on 1500 NM range which is about the maximum for the mission orbits selected. It has been assumed that the carrier lockon occurs at zero degrees elevation, and that data transmission would commence and terminate at elevation angles of 5° above the horizon. The system noise temperature was based on the cooled parametric amplifier receivers at 5° elevation, which are available at most of the MSFN sites. A tracking bandwidth between 50 and 700 Hz can be used, but to be conservative a 700 Hz was assumed. In order to assess error rate performance, it was assumed that multiple signals are precluded for these wideband channels. Thus no allowance is necessary for degradation due to interchannel interference. The equipment characteristics and operating procedures were obtained from data and information contained in Refs. C-1 to C-4.

Table C-1

SPACECRAFT TO MSFN TELEMETRY POWER BUDGET

Transmitter power (11.5 w)	10.6 db
Transmitter antenna gain	8.7
Transmitter and pointing loss	- 5.0
Receiver antenna gain (30')	44.0
Free space loss (2300 MHz, 1500 nm)	-168.3
Received power	-110.0 db

<u>Carrier</u>

Modulation loss ($\Delta\phi$ = 1.1 radian)	- 7. db
Carrier power	-117.
N_o (T = 126 °K)	-207.6
Carrier noise bandwidth (700 Hz)	28.5
Noise power	-179.1
Carrier to noise ratio	62.1
Required for carrier tracking	12.
Carrier margin	50.1 db

<u>Data</u>

Modulation loss ($\Delta\phi$ = 1.1 radian)	- 1.0 db
Signal power	-111.
Required ST/N_o (BT=1, PCM/PM, Pe - 10^{-6})	14.
Mechanization loss	- 2.
Data rate (20 mbps)	73
N_o (T = 126 °K)	-207.6
Threshold signal	-118.6
Data Channel margin	7.6

A margin of performance is necessary in communications channels due to possible equipment degradations. A margin of 3 db is a typical allowance for well defined links and due to the extensive experience in telemetry designs for Earth orbital applications, should be sufficient tolerance for this link. As can be seen from Table C-1, the carrier margin of 50.1 db and data channel margin of 7.6 db are greater than the necessary 3 db and therefore this link would perform adequately even under degraded conditions. The minimum transmitter power condition would be that which reduces the data channel margin to 3 db. Thus a power of 6 db (10.6 db minus 4.6 db), or 4 watts, is necessary for these wideband links. Table C-1 shows that a transmitter power of 11.5 watts is in excess of that required by the wideband channel and that additional system degradation could be tolerated and still maintain satisfactory performance under the assumed conditions.

REFERENCES - APPENDIX C

C-1. T. T. Tjhung and P. H. Wittice, "Carrier Transmission of Binary Data in a Restricted Band", IEEE Transactions on Communications Technology Vol Com-18, No. 4, August 1970, pp 295-304.

C-2. "Communication Systems Performance and Coverage Analysis for Apollo 11", Vol. IV MSC Internal Note No. EB-R-69-7, June 26, 1969.

C-3. "Manned Spaceflight Network User's Guide", GSFC Document MSFN No. 101.1, Goddard Space Flight Center, July 1970.

C-4. C. T. Dawson, et. al., "A Performance Analysis of the Apollo Unified S-Band Communications System for a Typical Lunar Mission", MSC Internal Note MSC-EB-R-67-1, May 1, 1967.

APPENDIX D

MSFN RECEIVER COVERAGE

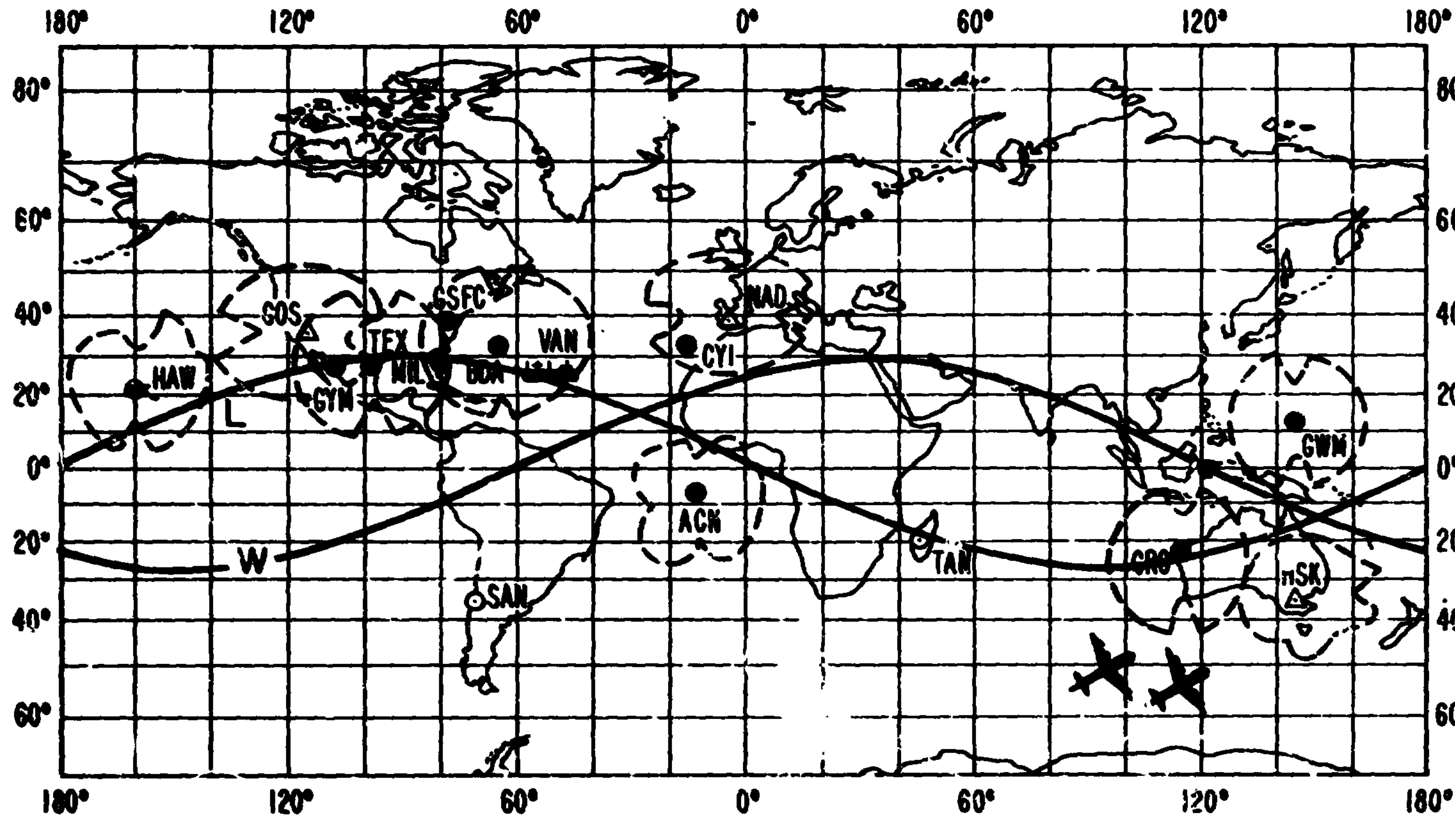
Figure D-1 shows the global locations of the ground installations of the MSFN, Ref. D-1. The coverage for these stations is also shown for a representative orbital altitude of 200 nm. At 245 nm, the postulated altitude of the interim lab, the radius of coverage would be about 10 percent greater. Note that the 85 foot dishes are mounted in such a way that their low angle blind zone, keyhole, lies East-West, while for the 30 foot dishes it lies North-South. This keyhole axis keyhole gives a better low angle coverage by the smaller dishes in support of lower inclination orbits, Ref. D-2.

Also shown in Figure D-1 are two ground traces for a 28.5° inclination orbit. The initial launch trajectory labeled "L" is repeatedly in view of a MSFN ground station. As the orbit regresses the spacecraft continues to pass over at least four such sites each orbit. The worst situation occurs after approximately 24 orbits (trace labeled "W") when coverage is at a minimum. At this time, only the Canary Islands (Cy I) station affords adequate coverage, and with brief contact indicated for Guam (GWM) and Madrid (MAD). This worst case situation could be remedied by relocation of the tracking ship USNS Vanguard, perhaps more near the Equator. It is stationed as shown on Figure D-1 for launch and orbital insertion coverage.

The higher inclination orbit (50°) displays a similar situation in regards to coverage. As shown in Figure D-2, for this inclination, the worst coverage (W) occurs with only marginal contact with Hawaii (HAW), Ascension Islands (ACN) and Santiago (SAN) if available. Again some supplementary coverage would be desired. For either of the mission inclinations, the availability of at least one MSFN site per orbit for data transmission is a conservative estimate. In practice several sites will be overflown for most orbits.

Most of the MSFN stations have dual receivers as is indicated in Table D-1. Thus these stations are capable of reception of four separate

LOCATION AND COVERAGE OF MSFN STATIONS
ORBIT INCLINATION OF 28.5°



POSITIONED AS REQUIRED

POSITIONED AS REQUIRED

○ PLANNED TO BE OPERATIONAL BY JULY-1972

● 30 ft DISH

△ 85 ft DISH

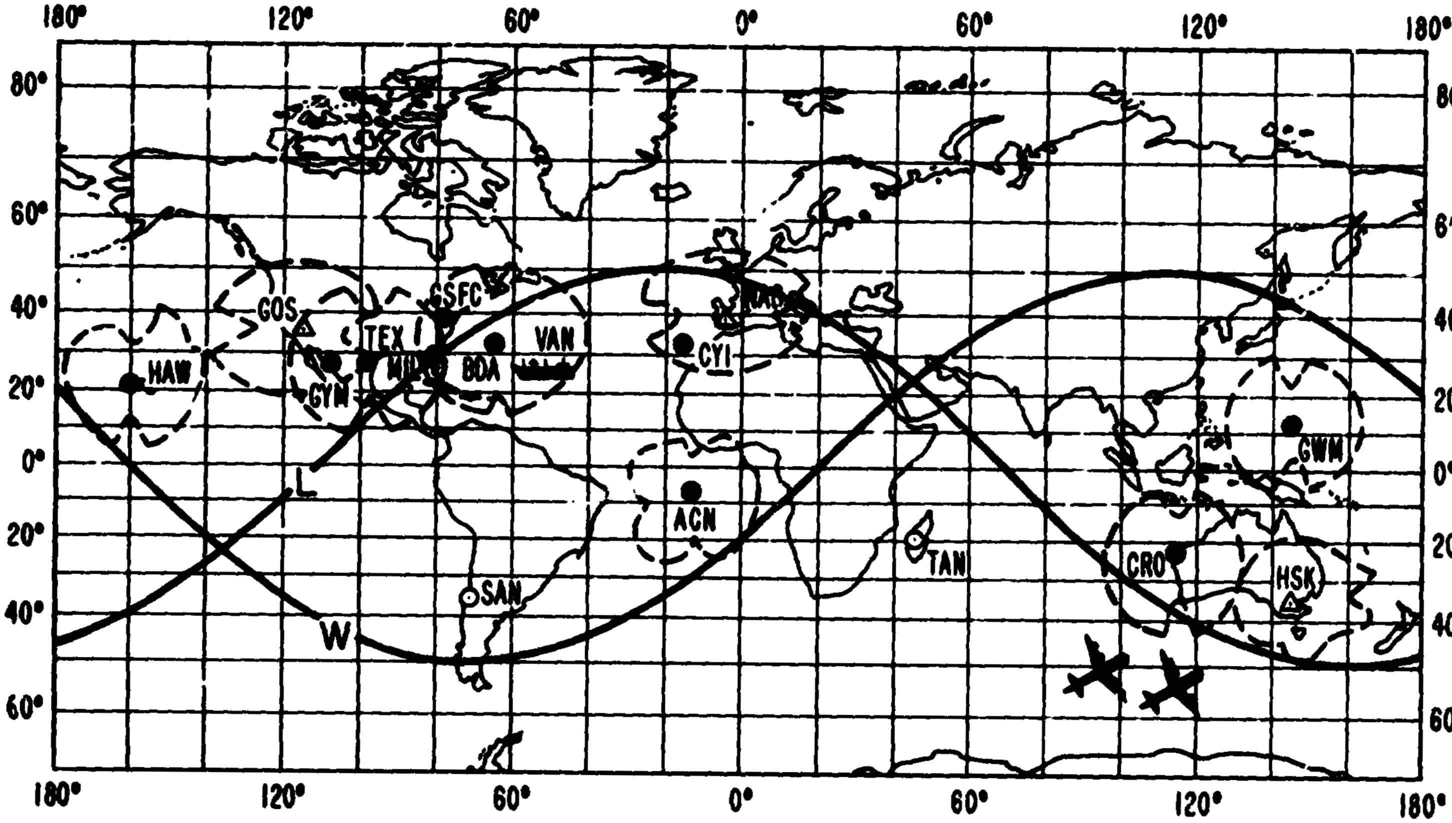
NOTE: RECEIVER RANGE SHOWN FOR ORBITAL ALTITUDE OF 200 n.mi.

L IS LAUNCH ORBIT

W IS POOREST TRANSMISSION ORBIT

Figure D-1.

LOCATION AND COVERAGE OF MSFN STATIONS
ORBIT INCLINATION OF 50°



POSITIONED AS REQUIRED

POSITIONED AS REQUIRED

○ PLANNED TO BE OPERATIONAL BY JULY-1972

● 30 ft DISH

△ 85 ft DISH

NOTE: RECEIVER RANGE SHOWN FOR ORBITAL ALTITUDE OF 200 n.mi.

L IS LAUNCH ORBIT

W IS POOREST TRANSMISSION ORBIT

Figure D-2.

Table D-1

EQUIPMENT SYSTEMS LOCATED AT MSFN STATIONS

System / Facility	USB Antenna for TLM, updata, Tracking, Voice, R&RR			VHF Antenna for TLM & Tracking				C-band Radar	High-speed Tracking Data	(VHF) A/G Voice	RSO Displays	High-speed TLM Data	UHF Command	S-band Telemetry[5]	136 MHz Telemetry	RSDPs[1]	SPAN	
	85' - Dual	30' - Dual	7' - USB	7' - Dish	AGAVE	TELTRAC	30' - Dish										Optical Telescope	Radio Telescope
ACN[4]		X				X			X	X		X	X	X	X	X		
BDA		X			X			X	X	X	X	X	X	X	X	X		
HSK	X								X			X		X		X		
CRO		X				X		X	X	X		X	X	X		X	X	X
CYI		X			X	X			X	X		X	X	X	X	X	X	X
GDS	X								X	X		X		X		X		
GWM		X				X			X	X		X	X	X	X	X		
GYM		X			X	X			X	X		X		X		X		
HAW		X			X	X		X	X	X		X	X	X		X		
MAD	X								X			X		X		X		
MIL		X				X			X	X		X		X		X		
SAN																		
TAN					X			X		X								
TEX		X			X	X			X	X		X	X	X		X		
ARIA 1-4[2]			X	X						X				X				
VAN		X					X	X	X	X		X	X	X	X	X		

[1] Apollo TLM and CMD RSDP's (642B).

[2] ARIA does not have updata or R&RR capability.

[3] Present configuration is 30' single station.

[4] ACN will have two separate 30' USB antennas and three range and range rate systems by March, 1971.

[5] Non-coherent S-band between 2260 and 2300 MHz.

[6] Planned to be operational by July 1, 1972.

REFERENCES - APPENDIX D

D-1. "Manned Space Flight Network User's Guide", GSFC Document MSFN No. 101.1, Goddard Space Flight Center, July 1970

D-2. "Manned Space Flight Network Antenna Coverage Data", GSFC Document X-820-67-581, Goddard Space Flight Center, November, 1967.

APPENDIX E
SPACECRAFT ORBIT MAINTENANCE

Consider that atmospheric drag has lowered the circular orbit altitude an amount Δr, and that a coplanar Hohmann transfer is made from the decayed orbit of radius $r-\Delta r$ back to the original orbit of radius r. The following calculates the velocity impulse required.

Let ΔV_1 = impulse on orbit of radius $r-\Delta r$ to raise orbit
ΔV_2 = impulse on orbit of radius r to circularize

$$\Delta V_1 = \left[\mu \left(\frac{2}{r-\Delta r} - \frac{1}{r-\frac{\Delta r}{2}}\right)\right]^{1/2} - \left[\frac{\mu}{r-\Delta r}\right]^{1/2}$$

$$= \left[\frac{\mu}{r} \left(\frac{2}{1-\frac{\Delta r}{r}} - \frac{1}{1-\frac{\Delta r}{2r}}\right)\right]^{1/2} - \left[\frac{\mu/r}{1-\frac{\Delta r}{r}}\right]^{1/2}$$

but for $\frac{\Delta r}{r} << 1$ we get

$$\Delta V_1 = \left\{ \frac{\mu}{r} \left[2\left(1 + \frac{\Delta r}{r}\right) - \left(1 + \frac{\Delta r}{2r}\right)\right]\right\}^{1/2} - \left[\frac{\mu}{r}\left(1 + \frac{\Delta r}{r}\right)\right]^{1/2}$$

then

$$\Delta V_1 = \sqrt{\frac{\mu}{r}} \left[\sqrt{1 + \frac{3}{2}\,\frac{\Delta r}{r}} - \sqrt{1 + \frac{\Delta r}{r}}\right]$$

again if $\frac{\Delta r}{r} << 1$

$$\Delta V_1 = \sqrt{\frac{\mu}{r}} \left[1 + \frac{3}{4}\,\frac{\Delta r}{r} - \left(1 + \frac{\Delta r}{2r}\right)\right]$$

$$\Delta V_1 = \sqrt{\frac{\mu}{r}}\,\frac{\Delta r}{4r}$$

and

$$\Delta V_2 = \sqrt{\frac{\mu}{r}} \left[1 - (1 - \frac{\Delta r}{2r})^{1/2}\right]$$

$$\Delta V_2 = \sqrt{\frac{\mu}{r}} \frac{\Delta r}{4r}$$

Therefore the total required impulse is

$$\Delta V = \Delta V_1 + \Delta V_2$$

$$\Delta V = \sqrt{\frac{\mu}{r}} \frac{\Delta r/r}{2}$$

where

$$V_c = \sqrt{\frac{\mu}{r}} = \text{circular velocity at radius } r$$

$$\Delta V = \frac{V_c}{2} \frac{\Delta r}{r}$$

but for Δr small we can write

$$\frac{\Delta P}{P} = \frac{3}{2} \frac{\Delta r}{r}$$

finally we can write

$$\Delta V = \frac{V_c}{3} \frac{\Delta P}{P}$$

APPENDIX F

LOGISTIC SPACECRAFT WEIGHTS

The detailed logistic spacecraft weights for both the stowed items and the fixed items as they were used for this report are given in this section. Tables F-1 and F-2 give the weights for the three-men and four-men versions of the Apollo command and service modules respectively. Tables F-3 and F-4 list the weights for the launch vehicle adapter and for the launch escape system.

Table F-1

APOLLO COMMAND MODULE WEIGHT SUMMARY

	Three Men	Four Men
Fixed Items	11,145 lbs.	12,035 lbs.
Structure	6,435	7,000
Stabilization and Control	200	200
Environmental Control System	570	650
Earth Landing System	690	750
Instrumentation	45	50
Electrical Power System	1,470	1,525
Reaction Control System	295	295
Communications	330	350
Controls and Displays	455	475
Crew Systems	260	340
Guidance and Navigation	395	400
Stowed Items	1,550	1,915
Crew & Crew Equipment	1,240	1,550
RCS Expendables	245	275
ECS Expendables	65	90
TOTAL	12,695	13,950

Table F-2

APOLLO SERVICE MODULE WEIGHT SUMMARY

	Three Men	Four Men
Fixed Items	11,260 lbs.	11,260 lbs.
Structure	3,955	3,955
Environmental Control System	255	255
Instrumentation	115	115
Electrical Power System	2,355	2,355
Main Propulsion System	3,595	3,595
Reaction Control System	945	945
Communications	40	40
Expendable Items	5,930	6,100
Reaction Control System	2,550	2,600
Propulsion System	2,725	2,800
Environmental Control System	135	150
Electrical Power System	520	550
TOTAL	17,190	17,360

Table F-3

LAUNCH VEHICLE ADAPTER WEIGHT SUMMARY

Saturn IV-B	
Structure	4095 lbs
Electrical Power System	60
Total	4155
Titan III-M	
Structure	4940 lbs
Electrical Power System	60
Total	5000

Table F-4

APOLLO LAUNCH ESCAPE SYSTEM WEIGHT SUMMARY

Structure	2445 lbs.
Electrical Power System	65
Propulsion System	2270
Propellants	3200
Q-Ball	25
Ballast	1240
Total	9245